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FLYING QUALITIES FLIGHT TESTING PHASE

CHAPTER 10
HIGH ANGLE OF ATTACK

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10.1 GENERAL INTRODUCTION TO HIGH ANGLE-OF-ATTACK FLIGHT

From the designer to the pilot, everyone associated with the flying qualities of high performance military aircraft, particularly the fighter or attack variety, is or should be aware of the importance of the high angle of attack flight regime. It is here that the aircraft will spend a significant amount of its time when performing the mission for which it was designed. It is here that the aircraft must display its most outstanding performance. It is also here that the aircraft, when pushed beyond its limits of controllability, can seemingly defy all laws of physics and principles of flight with which its surprised and often bewildered pilot is acquainted. The frequency of inadvertent loss of control at high angle of attack is such that many combat aircraft pilots are becoming firmly convinced that all pilots may be divided into two categories: those who have departed controlled flight, and those who will. Most thoroughly convinced are those pilots who fall into the former category.

The unfortunate fact concerning departure from controlled flight at high angle of attack is that many aircraft and pilots are lost each year due to failure to recover from the out-of-control flight condition. The circumstances surrounding the losses are varied. Departures from controlled flight may occur unintentionally during high-g maneuvers or intentionally during a nose-high deceleration to zero airspeed in an attempt to gain an advantage over an opponent in combat maneuvering; the aircraft may spin and the gyration be identified too late for recovery or a steep spiral may be mistakenly identified as a spin, causing recovery controls to be misapplied. Whatever the circumstances, departures from controlled flight result all too often in catastrophe (10.1:1). For this reason, test pilots in particular must be familiar with every facet of the high angle-of-attack flight regime.

10.2 INTRODUCTION TO STALLS

A broad definition of stall speed is the minimum steady speed attainable, or usable, in flight. A sudden loss of lift occurring at a speed just below that for maximum lift is considered the "conventional" stall, but this characterization is normally confined to high camber, high aspect ratio, straight wings. With more modern aircraft, it increasingly common for the minimum speed to be defined by some other characteristic, such as a intolerable buffet, an undesirable attitude, loss of control about any axis, or a deterioration of handling qualities.

For rather obvious safety and operational reasons, determination of stall characteristics is a first-order-of-business item in flight testing a new aircraft. Stall speeds are also required early in the test program for the determination of various test speeds.

10.2.1 SEPARATION

Separation, a condition wherein the streamlines fail to follow the body contours, produces a large disturbed wake behind the body and results in a pressure distribution greatly different from that of attached flow. On an aircraft, these changes in turn may produce:

a. A loss of lift (Figure 10.1)

b. An increase in drag

c. Control problems due to:

   1. Control surfaces operating in the disturbed wake

   2. Changes in the aerodynamic pitching moment due to a shift in the center of pressure and an altered downwash angle.

d. A degradation of engine performance

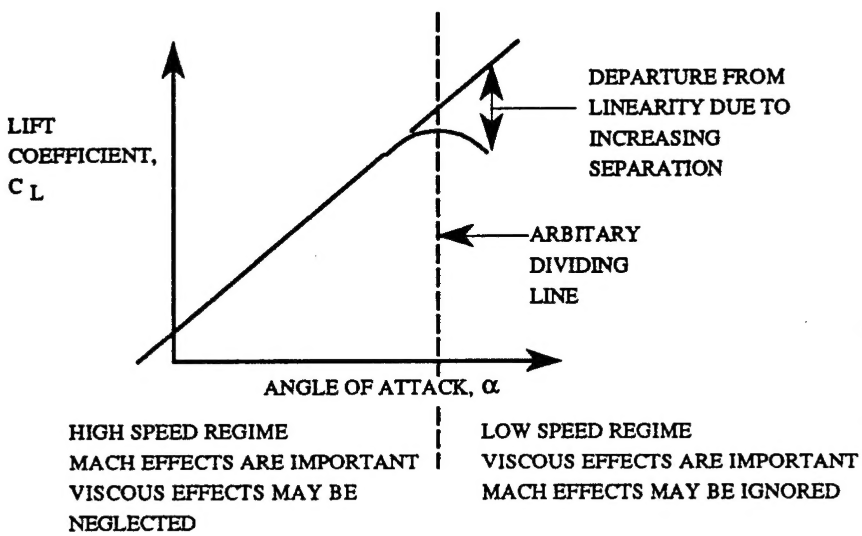

FIGURE 10.1. SEPARATION

Separation occurs at a point where the boundary layer kinetic energy has been reduced to zero, therefore the position and amount of separation is a function of the transport of energy into and out of the boundary layer and dissipation of energy within the boundary layer.

Some factors which contribute to energy transport are:

a. Turbulent (non-laminar) flow: Higher energy air from upper stream tubes is mixed into lower stream tubes. This type flow, characterized by a full velocity profile, occurs at high values of Reynolds (R_e) and involves microscopic turbulence.

b. Vortex generators: These devices produce macroscopic turbulence to circulate high energy air down to lower levels.

c. Slats and Slots: These devices inject high energy air from the underside of the leading edge into the upper surface boundary layer.

d. Boundary Layer Control: The blowing type of Boundary Layer Control (BLC) injects high energy air into the boundary layer while the suction type removes low energy air.

Two examples of energy dissipation functions are:

a. Viscous friction: Energy loss varies with surface roughness and distance traveled.

b. Adverse pressure gradient: Boundary layer energy is dissipated as the air moves against the adverse pressure gradient above a cambered airfoil section. The rate of energy loss is a function of.

   1. Body contours - Camber, thickness distribution, and sharp leading edges are examples.

   2. Angle of attack - Increased angle of attack steepens the adverse pressure gradient.

Some typical coefficient of lift versus angle of attack (C_L versus α) curves illustrating these effects are shown in Figure 10.2.

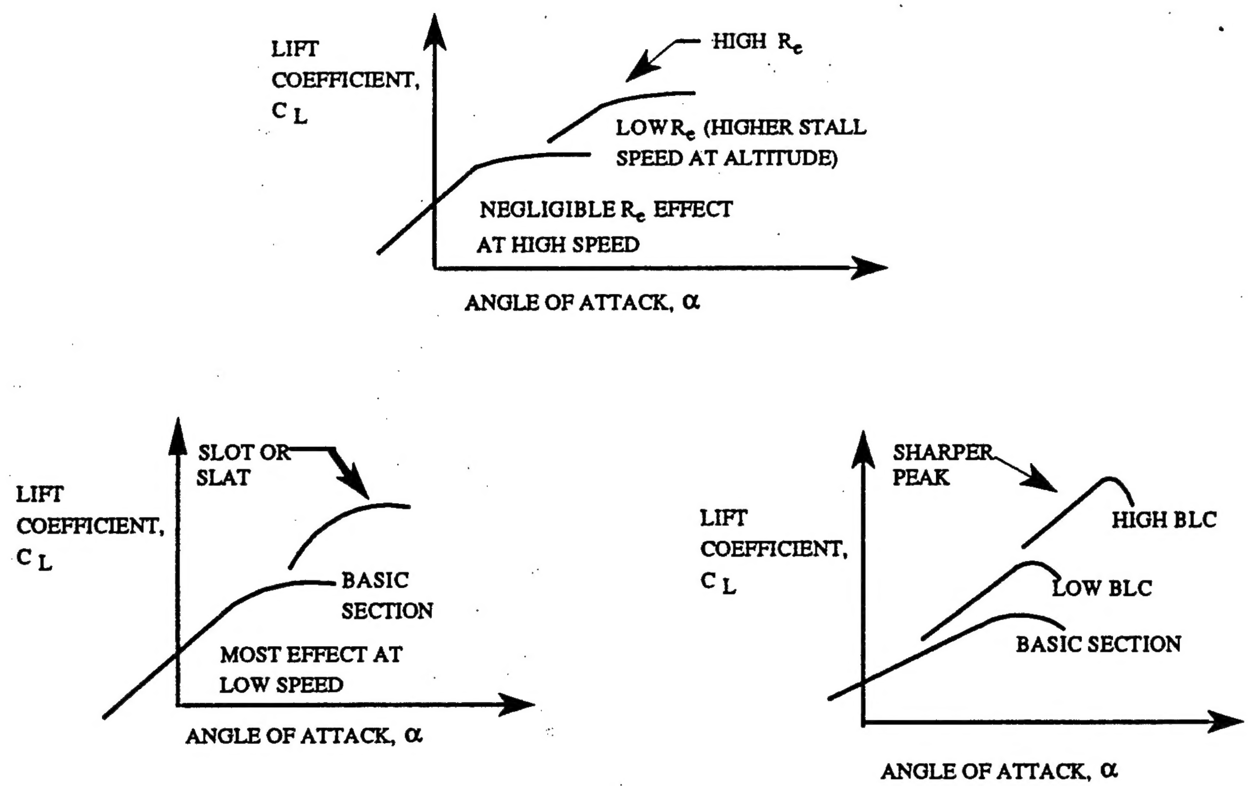

FIGURE 10.2. SEPARATION EFFECTORS

10.2.2 THREE-DIMENSIONAL EFFECTS

A three-dimensional wing exhibits aerodynamic properties considerably different from those of the two-dimensional airfoil sections of which it is formed. These differences are related to the planform and the aspect ratio of the wing.

10.2.3 PLANFORMS

Downwash, a natural consequence of lift production by a real wing of less than infinite span, reduces the angle of attack at which the individual wing sections are operating (Figure 10.3).

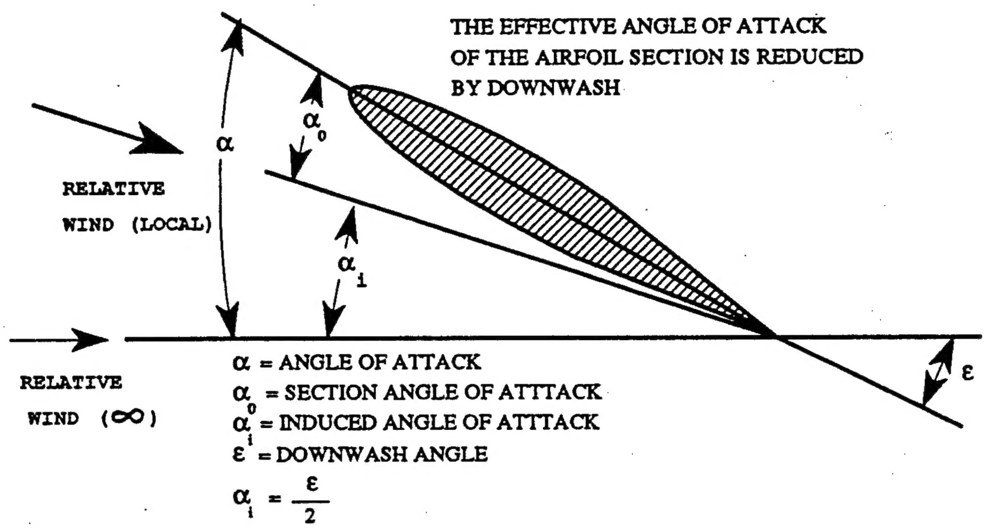

FIGURE 10.3. DOWNWASH EFFECT ON ANGLE OF ATTACK

An elliptical wing has a constant value of downwash angle along its entire span. Other planforms, however, have downwash angles that vary with position along the span. As a result, the lift coefficient for a particular wing section may be more or less than that of nearby sections, or that of the overall wing. Airfoil sections in areas of light downwash will be operating at high angles of attack, and will reach stall first. Stall patterns therefore depend on the downwash distribution, and vary predictably with planform as shown in Figure 10.4.

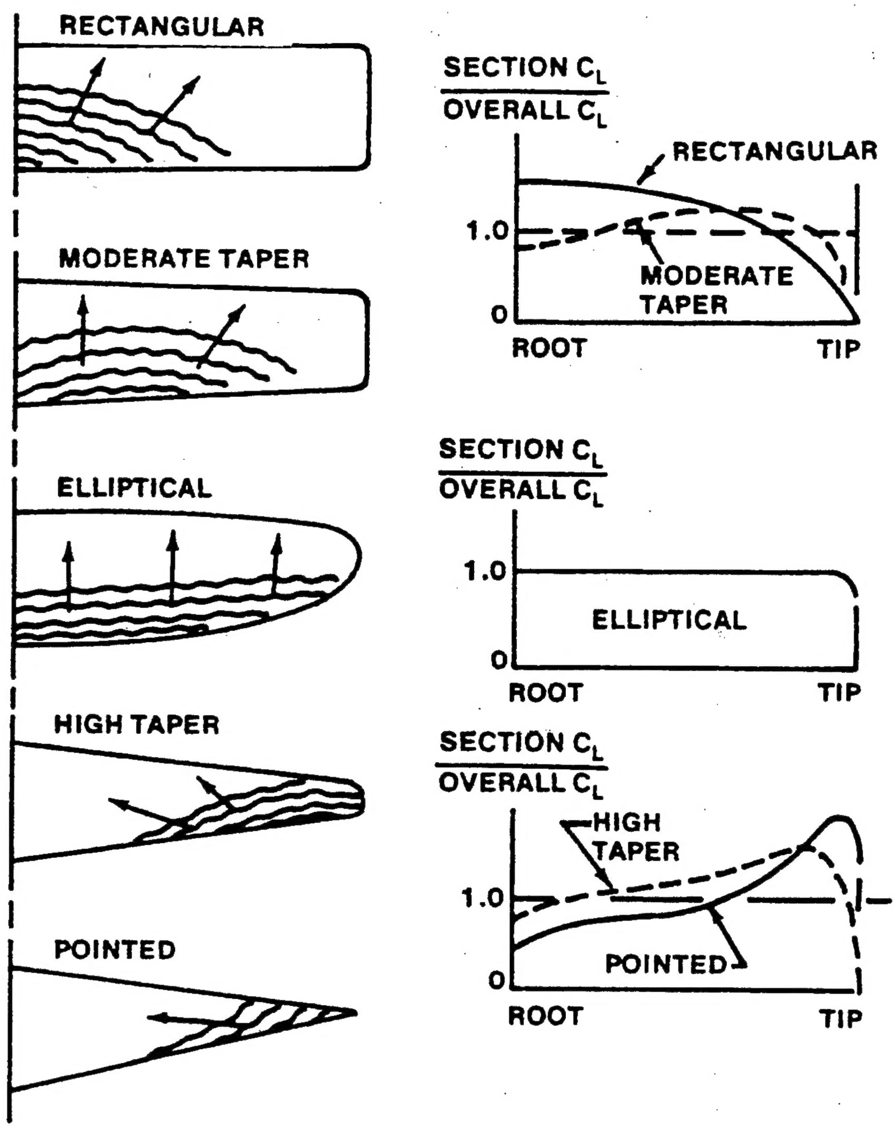

FIGURE 10.4. STALL PATTERNS

Sweptback and delta planforms suffer from an inherent spanwise flow (Figure 10.5). This is caused by the outboard sections being located to the rear, placing low pressure areas adjacent to relatively high pressure areas.

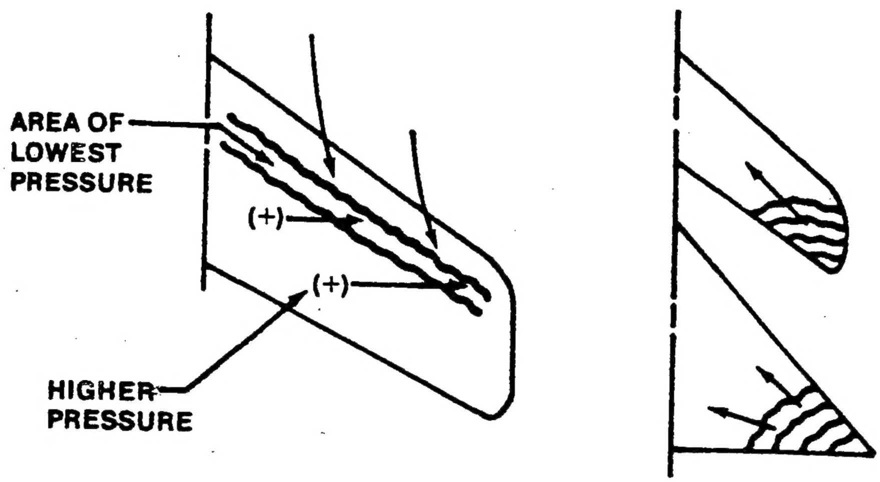

FIGURE 10.5. SPANWISE FLOW

This spanwise flow transports low energy air from the wake of the forward sections outboard toward the tips, inviting early separation. Both the sweptback and delta planforms display tip-first stall patterns.

Pointed or low chord wing tips are unable to hold the tip vortex, which moves further inboard with increasing angle of attack, as shown in Figure 10.6.

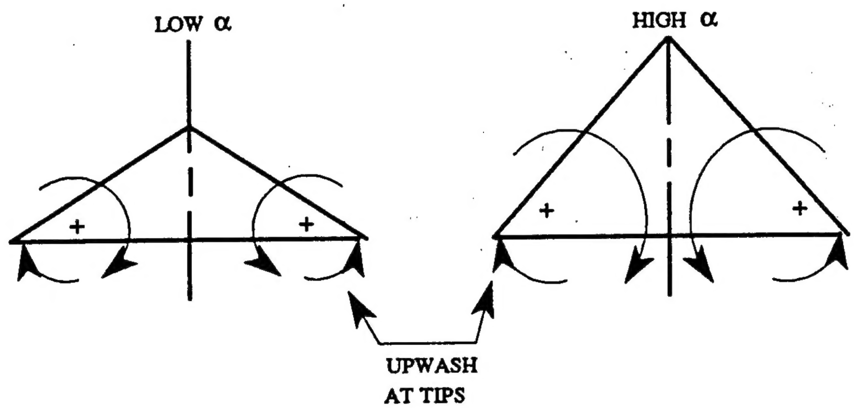

FIGURE 10.6. TIP VORTEX EFFECTS

The extreme tips operate in upwash and in the absence of aerodynamic fixes such as twist or droop, are completely stalled at most angles of attack.

10.2.4 ASPECT RATIO

Aspect ratio may be considered an inverse measure of how much of the wing is operating near the tips. Wings of low aspect ratio (much of the wing near the tip) require higher angles of attack to produce a given lift.

The curves shown in Figure 10.7 illustrate several generalities important to stall characteristics. High aspect ratio wings have relatively steep lift curve slopes with well defined peaks at C_{Lmax}. These wings have a relatively low angle of attack (and hence pitch angle) at the stall, and are usually characterized by a rather sudden stall break.

Low aspect ratio wings display the reverse characteristics: high angle of attack (high pitch angles) at slow speeds and poorly defined stalls. They can frequently be flown in a high sink rate condition to the right of C_{Lmax}.

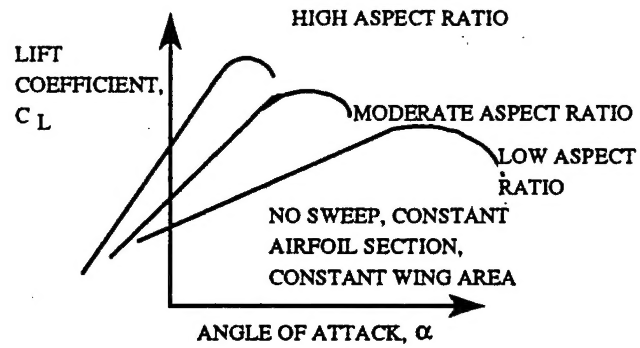

FIGURE 10.7. ASPECT RATIO EFFECTS

10.2.5 AERODYNAMIC PITCHING MOMENT

On almost all planforms the center of pressure moves forward as the stall pattern develops, producing a noseup pitching moment about the aircraft center of gravity (cg).

This moment is not great on most straight wing planforms and the characteristic root stall of these wings adds a compensating nosedown moment such that a natural pitchdown tendency exists at high angles of attack. This occurs because the stalled center section produces much less downwash in the vicinity of the horizontal tail, decreasing its download. If the tail actually enters the turbulent wake, the nosedown moment may be further intensified due to a decrease in elevator effectiveness. This latter case usually provides a natural stall warning in the form of an airframe and control buffet.

On swept-wing and delta planforms the moment produced by the center of pressure (cp) shift is usually more pronounced and the moment contributed by the change in downwash at the tail is noseup. This occurs because the wing root section remains unstalled, producing greater lift and greater downwash as the angle of attack increases. The inboard movement of the tip vortex system also increases the downwash behind the center of the wing. Horizontal tails, even in the vicinity of this increased downwash, will produce more download. If the tail is mounted such that it actually enters the downwash area at high angles of attack, such as on the F-101, an uncontrollable pitchup may occur.

Many fixes and gimmicks have been used to alter lift distribution and stall patterns. Tip leading edge extensions, tip slots and slats, tip washout and droop, fences and root spoilers are but a few. Horizontal tail position has also been adjusted as was necessary on the F-4C.

10.2.6 LOAD FACTOR CONSIDERATIONS

The relationship between load factor (n) and velocity may be seen on a V-n diagram, as shown in Figure 10.8. Every point along the lift boundary curve, the position of which is a function of gross weight, altitude, and aircraft configuration, represents a condition of C_{Lmax} (neglecting cases of insufficient elevator power). It is important to note that for each configuration, C_{Lmax} occurs at a particular α_{max}, indepenedent of load factor, i.e., an aircraft stalls at the same angle of attack and C_L in accelerated flight, with n = 2.0, as it does in unaccelerated flight, with n = 1.0. The total lift (L) at stall for a given gross weight (W) varies with load factor since L = nW. The increased lift at the accelerated stall must be obtained by a higher dynamic pressure (q).

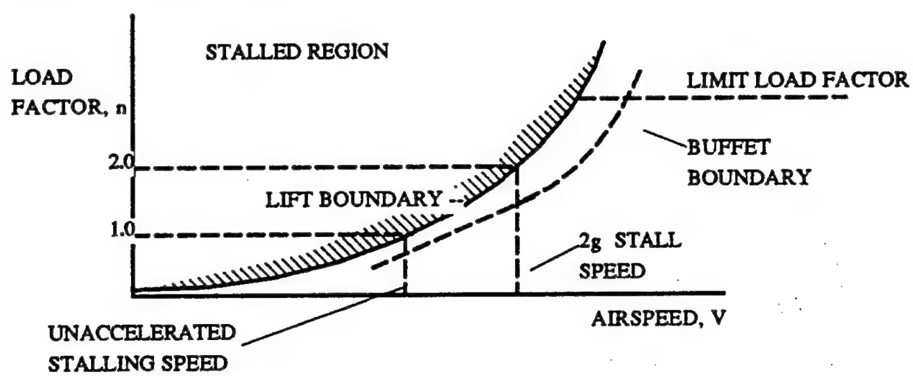

FIGURE 10.8. LOAD FACTOR EFFECTS

Thus stall speed is proportional to the square root of n, making accurate control of normal acceleration of primary importance during stall tests.

10.3 INTRODUCTION TO SPINS (10.2: 1-1, 1-2)

The early glider flights of the Wright brothers often ended by dropping off on one wing, out of control, with a wingtip eventually striking the Kitty Hawk, North Carolina, sand in a rotary motion. While the low altitude of these flights prevented motion from developing fully, it seems clear that these were departures into incipient spins.

In those early days of manned flight the spin was as dangerous as it is today. When the Wright brothers first tried warping the wings to roll into a turn, they found that the banking was accompanied by a dangerous tendency to diverge in yaw at high angle of attack. Adding a fixed vertical fin helped stabilize the 1902 glider, but the loss-of-control problem persisted. Orville Wright reasoned that a hinged vertical rudder could produce a counter yawing moment to keep the yaw from starting and thus enable the flyer to remain under control. This was tried first with rudder deflection connected to the wing-warp control, then with the pilot controlling the rudder separately. The fix was effective but required the pilot's constant attention. Proper spin recovery controls were not generally known until 1916, when flight test experiments on spin recovery procedures were conducted at the Royal Aircraft Factory, Farnborough, in an FE-8. Wing Commander Macmillan is responsible for a fascinating early history of the spin in Aeronautics, 1960-62 issues.

For early airplanes the spin recovery technique was at least rational if not instinctive: forward stick and rudder opposing the yawing motion should stop the rotation and unstall the wing. Once these recovery controls were known, World War I pilots used the spin as a maneuver to lose altitude without gaining airspeed. Then in the 1920's some of the more peculiar spin modes were recognized as problems. Accident summaries from that era show spins were involved in about three percent of all accidents reported and in twenty to thirty percent of the fatal accidents.

Analytical studies and dynamic wind tunnel testing to reduce the stall/spin problem were reported in England as early as 1917. Autorotation was observed in the wind tunnels, and the first analytical prediction methods were developed by Glauert and Lindemann. Gates and Bryant presented a comprehensive survey of spinning in 1927. About 1930, a method of determining the flight path and attitude of a spinning aircraft was put into use. Rotation rates about and accelerations along the principal axes, as well as vertical velocity were

measured and recorded photographically. This information was used to define the motion of the aircraft and could then be used in conjunction with the analytical prediction methods.

In the 1920's and 1930's, several forms of testing were being performed. One safety measure used in full-scale testing was to attach external ballast, that when released would cause the center of gravity of the airplane to move forward, thus returning the airplane to a controllable configuration. Because of the hazards involved in stall/spin flight testing, researchers hesitant to use full-scale aircraft tried free-flight models. One of the early spin models was dropped from the top of a 100-foot balloon hangar at Langley Field. This proved an inadequate means of obtaining data, and soon vertical wind tunnels were being built to investigate spinning (1930 in the United States, 1931 in England). In 1945, the U.S. Army Air Force dropped an instrumented model from a Navy blimp to study spin entry and recovery.

As jet aircraft were developed, the inertial characteristics of fighters in particular were changed to the point that spins and other post-stall motions became more troublesome and even required different recovery techniques. Most stall/spin problems were identified by the Wright Air Development Center Spin Symposium; some analysis methods had been developed, and the electronic digital computer provided a useful tool with which to examine the stall/spin problem. Then suddenly the emphasis was shifted to space. With little management interest and rather poor expectations of improvement, resources for stall/spin research were quite limited. Our Air Force tended to concentrate on performance improvements, which have often aggravated stability and control problems at high angles of attack. Today, a large and costly Air Force accident record and a renewed emphasis on maneuver capability have led to a concerted effort to solve the problems associated with aircraft operating in the stall/spin flight regime.

Large aircraft have also experienced stall/spin problems. For example, several B-58s were lost in spins. Automatic trimming of the control-stick force was mechanized in such an insidious way that an inattentive pilot might not be aware of a slowdown to stall speed. Trouble with fuel management could result in an extreme aft center of gravity, at which B-58 stability and control were deteriorated. The C-133, on long flights, would climb to an altitude approaching its absolute ceiling. Poor stall warning and a vicious stall while trying to fly there are thought to have caused the disappearance of several C-133 aircraft. It has

become customary to require analysis and spin tunnel testing of all U.S. military airplanes even though flight demonstration of large, low-maneuverability types is limited to stalls with only moderate control abuse.

The military specification for flying qualities defines good high angle of attack characteristics in terms that are qualitative rather than quantitative. The airplane must exhibit adequate stall warning, and in addition the stall must be easily recoverable. We now emphasize resistance to violent departures from controlled flight, while retaining requirements for recovery from attainable post-stall motions. The definitions of good high angle of attack characteristics will differ for the various classes of aircraft; but with respect to fighter aircraft, a pilot should not have to worry about loss of control while flying within his useful maneuver envelope. We shall need quantitative requirements that will be of more use in the design stage for all classes of airplanes.

Generally post-stall design and testing have emphasized spins and spin recovery, taking the point of view that assurance of recoverability from the worst possible out-of-control situation guarantees safety. This philosophy falls short in several respects. Resistance to departure has not been emphasized adequately. The motions can be disorienting, and recovery control inputs such as ailerons with the spin are unnatural. And as airplanes grow larger and heavier, altitude loss becomes excessive. F-111 instructions, for example, are to eject if spin recovery has not commenced upon reaching 15,000 feet altitude. Spins and spin recovery should not be neglected, but emphasis needs to shift to departure resistance and early recovery.

10.3.1 DEFINITIONS

10.3.1.1 STALL VERSUS OUT-OF-CONTROL

Stalls and associated aerodynamic phenomena have been previously described, but it is worth repeating the formal definition of stall angle of attack from page 7 of MIL-STD-1797A FLYING QUALITIES OF PILOTED AIRCRAFT (10.3). In terms of angle of attack, the stall is defined as the lowest of the following:

a. Angle of attack for the highest steady load factor normal to the flightpath that can be attained at a given speed or Mach.

b. Angle of attack, for a given speed or Mach, at which uncommanded pitching, rolling, or yawing occurs (4.8.4.2). Angular limits of 20° (Classes I, II, or III) or 30° (Class IV) are recommended in Paragraph 4.8.4.2.2 of MIL-STD 1797A.

c. Angle of attack for a given speed or Mach, at which intolerable buffeting is encountered.

d. An arbitrary angle of attack allowed by Paragraph 4.1.4.3 of MIL-STD-1797A, which may be based on such considerations as ability to perform attitude corrections, excessive sinking speed, or ability to execute a go-around.

MIL-F-83691B FLIGHT TEST DEMONSTRATION REQUIREMENTS FOR DEPARTURE RESISTANCE AND POST-DEPARTURE CHARACTERISTICS OF PILOTED AIRPLANES uses the same definition excluding d. above. Its predecessor MIL-S-83691A defined the stall angle of attack more simply: the angle of attack for maximum usable lift at a given flight condition. This latter definition is the one most useful in this course, but the student must understand that "maximum usable lift" is determined from one of the four conditions given above.

10.3.1.2 DEPARTURE

Departure is defined as "that event indicating loss of control which may develop into a post departure gyration (PDG), spin, or deep stall condition (MIL-F-83691B, Paragraph 6.3.10). The departure may be characterized by divergent, large-amplitude, uncommanded aircraft motions, such as nose slice or pitch-up." An AOA excursion (AOA momentarily exceeds the limiter AOA but does not result in uncommanded aircraft motions) is not considered a departure. Notice two things about this definition. First, departure is an event that has no direct correlation to an angle of attack. According to this definition the airplane can technically depart although stall angle of attack has not been reached. (Author's note: this philosophy of separating departure from stall angle of attack is contrary to the definition of stall angle of attack contained in MIL-STD-1797A which includes "angle of attack, for a given speed or Mach, at which uncommanded pitching, rolling, or yawing occurs." More about this later in the definitions of post-stall gyrations and post-departure gyrations.) The second

point is that only one of three motions may result after departure - the aircraft enters either a PDG, spin, or deep stall (of course, a PDG can progress into a spin or deep stall). Implicit in this definition is the implication that an immediate recovery cannot be attained. For example, a light aircraft whose stall is defined by a g-break, may recover immediately if the longitudinal control pressure is relaxed. However, note that movement or position of controls is not mentioned in the definition. The same light aircraft that would not depart if control pressures were relaxed at the stall may depart and enter a spin if pro-spin controls are applied at the stall. Hence, in discussing susceptibility or resistance to departure one must specify control positions as well as loading and configuration.

As mentioned in the definition, the departure event is usually a large amplitude, uncommanded, and divergent motion. Such descriptive terms as nose slice or pitch-up are commonly used to describe the event. Large amplitude excursions imply changes in yaw, roll, or pitch greater than 20° (Class I, II, and III) or 30° (Class IV) (MIL-STD-1797A, Appendix A, paragraph 4.8.4.2.2). Uncommanded motions are motions not intended by the pilot, even though the control positions are legitimately causing the departure. The aircraft may not follow the pilot's commands for a number of reasons: the high angle of attack may render the control surface ineffective when moved to its desired position; or the pilot may be unable to position the stick to put the surface in the desired position due to lateral or transverse g loads. In either of these conditions the aircraft motion is "uncommanded." Finally, a divergent motion is one which either continuously or periodically increases in amplitude. The T-33 usually exhibits a "bucking" motion after the stall in which the nose periodically rises and falls. However, the motion is not divergent unless aggravated by full aft stick or some other pro-spin control. The T-38 will sometimes exhibit a non-divergent lateral oscillation (wing rock) near the stall angle of attack. Neither of these motions are normally considered departures, though their occurrence does serve as warning of impending departure if further misapplications of controls are made. With this sort of background it is easy to see why a departure is so hard to define, yet is relatively easy for a pilot to recognize. Next, one must examine the terms "post-departure gyration", "post-stall gyration," "spin" and "deep stalls."

10.3.1.3 POST-DEPARTURE GYRATION AND POST-STALL GYRATION

A post-departure gyration is an uncontrolled motion about one or more airplane axes following departure (MIL-F-83691B, Paragraph 6.3.12). PDG is a very difficult term to define concisely because it can occur in so many different ways. Frequently, the motions are completely random about all axes and can only be described as post-departure gyrations. On the other hand, such terms as snap roll, rolling departure, or tumble may be appropriate as long as they imply a PDG. The main difficulty lies in distinguishing between a PDG and either the incipient phase of a spin or an oscillatory spin. The PDG is differentiated from a spin by the lack of a predominant sustained yawing motion. The PDG is differentiated from a deep stall by the presence of significant angular motions and accelerations. A post-stall gyration (PSG) is defined as a post-departure gyration occurring in the post-stall flight regime (MIL-F-83691B, Paragraph 6.3.13). While a PSG involves angles of attack higher that the stall angle, lower angles may be encountered intermittently in the course of the motion. What MIL-F-83691B is trying to do is differentiate between departures beyond "stall AOA" and those prior to "stall AOA". An example of a departure prior to stall AOA would be an F-15 autoroll which typically occurs at a very low angle of attack. Remember, however, that this philosophy is contrary to the definition of stall AOA contained in MIL-STD-1797A.

10.3.1.4 SPIN

A spin is a sustained yaw rotation at angles of attack above stall (MIL-F-83691B, Paragraph 6.3.14). By definition, a spin requires two conditions - a yaw rate and a stall. This definition bears a bit of explanation in that a spin is certainly not limited to only a yaw rotation. Only the perfect flat spin ($\alpha = 90°$) could satisfy that constraint. The inference is, however, that the yaw rotation is dominant in characterizing a spin. Indeed, to a pilot, the recognition of a sustained (though not necessarily steady) yaw rate is probably the most important visual cue that a spin is occurring. Even though roll rate and yaw rate are often of nearly the same magnitude, the pilot still ordinarily recognizes the spin because of the yaw rate. In steep spins (with α relatively close to α_S), it is quite easy to confuse the roll rate and yaw rate and pilots sometimes have difficulty in recognizing this type of motion and treating it as a spin. The steep inverted spin is particularly confusing since the roll and yaw

rates are in opposite directions. Once again though, the yaw rate determines the direction of the spin and the required control manipulations to recover. All in all, it is well to remember that the spin is truly a complicated maneuver involving simultaneous roll, pitch, and yaw rates and high angles of attack. Even though the overall rotary motion in a spin will probably have oscillations in pitch, roll, and yaw superimposed upon it, it is still most easily recognized by its sustained yawing component.

10.3.1.5 DEEP STALL

A deep stall is an out-of-control flight condition in which the airplane is sustained at an angle of attack beyond that for stall or limiter AOA while experiencing negligible rotational velocities (MIL-F-83691B, Paragraph 6.3.15). It may be distinguished from a PSG by the lack of significant motions other than a high rate of descent. The deep stall may be a fairly stable maneuver such as a falling leaf, or it can be characterized by large amplitude angle of attack oscillations. Depending upon aircraft external store loading, center of gravity location, and flap/slat configuration, some low rate oscillatory yawing, rolling, or pitching motions may be present in a deep stall. Depending on the pitching moment coefficient, recovery may or may not be possible.

10.3.2 SUSCEPTIBILITY AND RESISTANCE TO DEPARTURES AND SPINS

Susceptibility/resistance to departures and spins has become an extremely important design goal for high performance aircraft. The report on the F-111 stall/spin prevention program (10.5) offers convincing proof that such design emphasis is overdue. But, for the designer to meet this requirement in an aircraft and for the test pilot to test against this requirement, it is essential that the words "susceptible" and "resistant" be understood alike by all concerned.

The degree of departure/spin/deep stall resistance for all Classes of airplanes is determined by the test phase in which departures/spins/deep stalls first occur while performing those maneuvers listed in Tables I or II in MIL-F-83691B. These tables describe the phases of testing and types of maneuvers required based on the airplane Class. Table I applies to

airplanes without AOA limiting devices and Table II addresses airplanes with AOA limiting devices. Susceptibility/resistance classification is in accordance with Table III in the MIL-F-83691B which is summarized here as Table 10.1.

TABLE 10.1 SUSCEPTIBILITY/RESISTENCE CLASSIFICATION

The airplane is classified as ______ to departure, spin, or deep stall if that event occurs in ______	Extremely Susceptible in Test Phase A Susceptible in Test Phase B Resistent in Test Phase C Extremely Resistent in Test Phase D

10.3.2.1 EXTREMELY SUSCEPTIBLE TO DEPARTURE, SPINS, OR DEEP STALLS (PHASE A)

An aircraft is said to be extremely susceptible to departure, spins, or deep stalls if that event occurs with the normal application of pitch control alone or with small roll, yaw, and decoupled control inputs. Decoupled control inputs are defined as unconventional controls such as direct normal force, direct sideforce, pitch pointing, yaw pointing, vertical translation, lateral translation, and flight path control using thrust vectoring (excluding V/STOL flight). The only allowable roll and yaw control inputs are those normally associated with a given maneuver task. In short, an airplane that departs or enters a spin during Phase A of the flight test demonstration falls within this category (MIL-F-83691B, Paragraph 6.3.16).

10.3.2.2 SUSCEPTIBLE TO DEPARTURE, SPINS, OR DEEP STALLS (PHASE B)

An aircraft is said to be susceptible to departure, spins, or deep stalls when the event generally occurs with the application or brief misapplication of pitch control and roll, yaw and decoupled controls that may be anticipated in normal operational use. The amount of misapplied controls to be used will be approved by the procuring activity for Phase B of the flight test demonstration. In other words, each aircraft will be stalled and aggravated control inputs will be briefly applied to determine departure, spin, deep stall susceptibility.

10.3.2.3 RESISTANT TO DEPARTURE, SPINS, OR DEEP STALLS (PHASE C)

An aircraft is said to be resistant to departure, spins, or deep stalls if the event occurs only with a large and reasonably sustained misapplication of controls. "Reasonably sustained" means up to three seconds before recovery is initiated (MIL-F-83691B, Table 1). This time delay may be increased for aircraft without positive indication of impending loss of control.

10.3.2.4 EXTREMELY RESISTANT TO DEPARTURE, SPINS, OR DEEP STALLS (PHASE D)

An aircraft is said to be extremely resistant to departure, spins, or deep stalls if these motions occur only after an abrupt and inordinately sustained application of gross, abnormal, pro-departure controls. Aircraft in this category will only depart, spin, or deep stall in Phase D of the flight test demonstration when the controls are applied and held in the most critical manner to attain each possible mode of post-stall motion and held for various lengths of time up to 15 seconds or three spin turns, whichever is longer.

10.3.3 THE MECHANISM OF DEPARTURE (10.6)

As an aircraft approaches stall conditions the aerodynamic changes produced by flow breakdown over the wing and tail result in degraded stability and control effectiveness. It is the extent of this degradation that determines whether the aircraft is departure-prone or departure-resistant. Use of controls to prevent departure may not be effective because both aileron and rudder effectiveness are greatly reduced in the stall and, in addition, adverse yaw characteristics may prohibit the use of ailerons.

If the aircraft becomes directionally unstable but still retains a stable dihedral effect of sufficient magnitude, its departure will not be divergent. When both directional stability and dihedral effect become unstable, then any disturbance such as a gust or control input can result in a departure. The departure may be self-terminating or it may result in subsequent spin entry. Once a departure has occurred, an important question arises: is there any restoring tendency which will impede further uncontrolled excursions and if so, how is it manifested?

If the aircraft has a pitch down at the stall, or if the longitudinal control retains full effectiveness, the departure may be terminated through reduction in angle of attack. There may even be a restoring tendency at large sideslip angles where the vertical tail emerges from

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the wing-body interference field to restore both directional stability and dihedral effect enough to attenuate the departure tendency.

When increasing angles of attack and sideslip both result in further degradations in directional stability and dihedral effect and longitudinal control is not effective (or is not applied), departure and subsequent spin entry are highly probable.

The stability derivatives C_{n_β} and C_{l_β} play an important role in high performance aircraft operating at high AOAs. As AOA is increased, C_{n_β} decreases to the point where it becomes negative; at this point, sideslip angles could start to diverge due to lack of directional stability. A stable dihedral effect (C_{l_β}) will normally tend to decrease sideslip angle by rolling into the direction of yaw (away from β). This roll-off tendency prevents a pure yaw divergence thereby providing a natural aerodynamic recovery from a departure.

Since aircraft react to combinations of C_{n_β}, C_{l_β} and kinematic as well as inertial coupling, a parameter which includes all of these effects is useful in predicting departure susceptibility. This parameter is called the Directional Departure Parameter ($C_{n_{\beta\ Dynamic}}$).

10.3.3.1 DIRECTIONAL DEPARTURE PARAMETER

The directional departure parameter represents an aircraft's directional stability with respect to its flight path rather than its body axes. Consider an aircraft body and stability axes as shown in Figure 10.9.

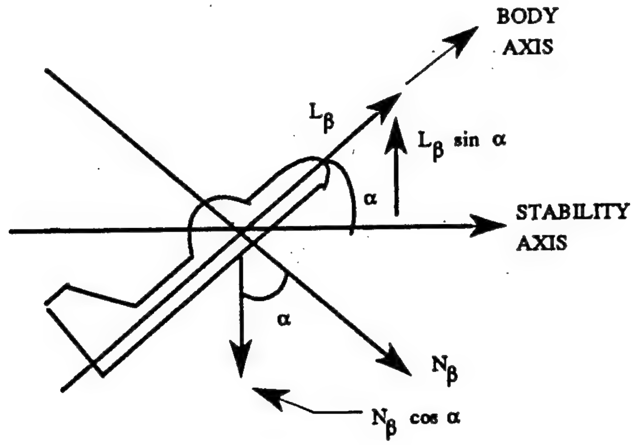


FIGURE 10.9. STABILITY AXIS RESOLUTION

Resolving the body axes rolling and yawing moments (in terms of dimensional stability derivatives) into the stability axes $_s$, we get

$$N_{\beta s} = N_\beta \cos\alpha - L_\beta \sin\alpha \tag{10.3}$$

Recalling that the normalizing factors for each dimensional derivative are

$$N_\beta = \frac{1}{I_z} C_{n_\beta} \qquad\qquad L_\beta = \frac{1}{I_x} C_{l_\beta}$$

and substituting these relationships into Equation 10.3

$$\frac{1}{I_z} C_{n_{\beta_s}} = \frac{1}{I_z} C_{n_\beta} \cos\alpha - \frac{1}{I_x} C_{l_\beta} \sin\alpha$$

multiplying by I_z we get

$$C_{n_{\beta_{DYN}}} = C_{n_\beta} \cos \alpha - \frac{I_z}{I_x} C_{l_\beta} \sin \alpha \qquad 10.4$$

where

$C_{n_{\beta_{DYN}}}$ = Directional departure parameter - stability axes

C_{n_β} = Directional stability derivative - body axes

C_{l_β} = Dihedral effect derivative - body axes

Normally a departure can be anticipated when $C_{n_{\beta DYN}}$ is negative. Notice that even if C_{n_β} goes negative, $C_{n_{\beta DYN}}$ may remain positive if the inertia ratio I_Z/I_X is high and if C_{l_β} is negative (stable). Most modern fighter aircraft have a high inertia ratio which is beneficial as long as C_{l_β} remains negative. However, once C_{l_β} becomes positive, the high inertia ratio will have an adverse effect on $C_{n_{\beta DYN}}$ and departure susceptibility. In general, a high negative (stable) C_{l_β} is desired for fighter type aircraft. For a given value of $C_{n_{\beta DYN}}$, if C_{l_β} is negative and large in magnitude, a roll divergence can be expected. A comparison of C_{n_β} and $C_{n_{\beta DYN}}$ for the A-7 and F-18 aircraft is shown in Figures 10.10 and 10.11, respectively.

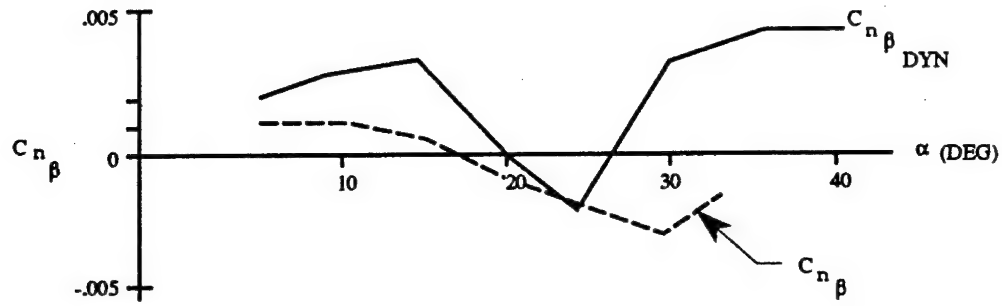


FIGURE 10.10. DIRECTIONAL STABILITY, A-7

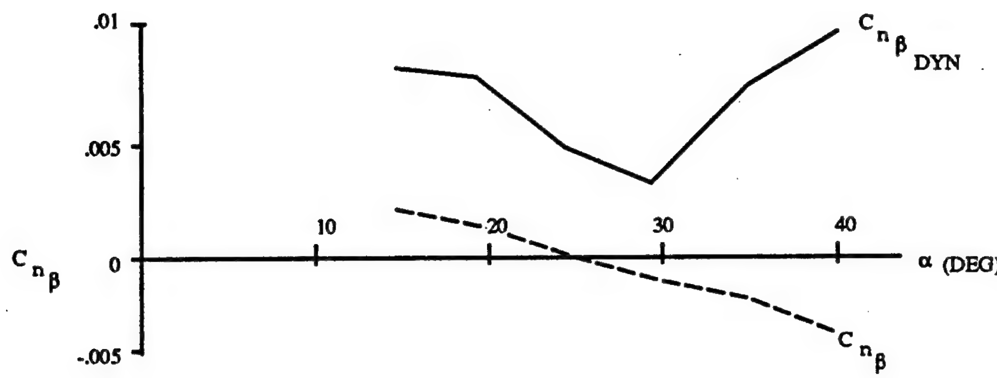


FIGURE 10.11. DIRECTIONAL STABILITY, F-18

10.3.3.2 LATERAL CONTROL DIVERGENCE PARAMETER (LCDP)

The lateral control divergence parameter (also referred to as the lateral control departure parameter) has been used in various forms for different uses. It is developed from the simplified rolling and yawing moment equations assuming the aircraft to be laterally and directionally trimmed. The lateral control divergence parameter predicts at what angle of attack roll reversal is expected to occur when using ailerons. Reversal in this sense does not mean aeroelastic aileron reversal, but that condition where the rolling moment due to sideslip,

resulting from adverse yaw, overpowers the rolling moment commanded by the ailerons. To the pilot this condition is known as reverse aileron command or roll reversal.

This parameter has been correlated with spin entry tendency due to aileron adverse yaw for several fighter aircraft. The expression for LCDP also contains sideslip derivatives and is

$$\boxed{LCDP = C_{n_\beta} - \frac{C_{n_{\delta_a}}}{C_{l_{\delta_a}}} C_{l_\beta}} \qquad 10.5$$

Aileron reversal occurs when LCDP becomes negative. Note the similarity of the role that dihedral effect plays in this expression relative to its role in $C_{n_{\beta DYN}}$. It again acts as a "multiplier"and the benefit or detriment in this case depends upon the sign of $C_{n_{\delta a}}$. If the aircraft has adverse yaw due to aileron ($-C_{n_{\delta a}}$), then a stable dihedral effect ($-C_{l_\beta}$) will have an adverse effect. The opposite is true if the aircraft has proverse yaw. If the aircraft has an unstable dihedral effect, then the combined effects of LCDP and $C_{n_{\beta DYN}}$ will determine the type of departure following an aileron input.

10.3.4 SPIN MODES

Adjective descriptors are used to describe general characteristics of a given spin and these adjectives specify the spin mode. Average values of angle of attack, for example, would allow categorization of the spin as either upright (positive angle of attack) or inverted (negative angle of attack). An average value of angle of attack would also allow classification of a spin as either flat (high angle of attack) or steep (lower angle of attack). Finally, the average value of the rotational rate and the oscillations in angular rates about all three axes determines the rate and oscillatory character of the spin. One descriptive modifier from each of these groups may be used to specify the spin mode, see Table 10.3.

TABLE 10.3. SPIN MODE MODIFIERS

Sense	Attitude	Rate	Oscillations
Erect	Extremely steep	Slow	Smooth
Inverted	Steep	Fast	Mildly Oscillatory
	Flat	Extremely Rapid	Oscillatory
			Highly Oscillatory
			Violently Oscillatory

The most confusing thing about mode identificaction is the proper use of the attitude and oscillation modifiers. Perhaps the following tabulated data, Table 10.4, extracted from Reference 10.7, will provide insight into understanding how to use these terms.

TABLE 10.4. F-4E SPIN MODES

Mode	Average AOA (deg)	AOA Oscillations (deg)	Yaw Rate (deg/sec)	Roll Rate (deg/sec)	Pitch Rate (deg/sec)
Steep-Smooth	42	± 5	40-50	50	15
Steep-Mildly Oscillatory	45-60	± 10	45-60	—	—
Steep-Oscillatory	50-60	± 20	50-60 (with large oscillations)	Same as yaw rate	—
Flat-Smooth	77-80	Negligible	80-90	25	7

Note: One mode reported in Reference 10.7 has been omitted from this table because the terminology did not fully comform to that of Reference 10.4. It was called "highly oscillatory"with angle of attack excursions of 180°.

10.3.5 SPIN PHASES

A typical spin may be divided into the phases shown in Figure 10.12.

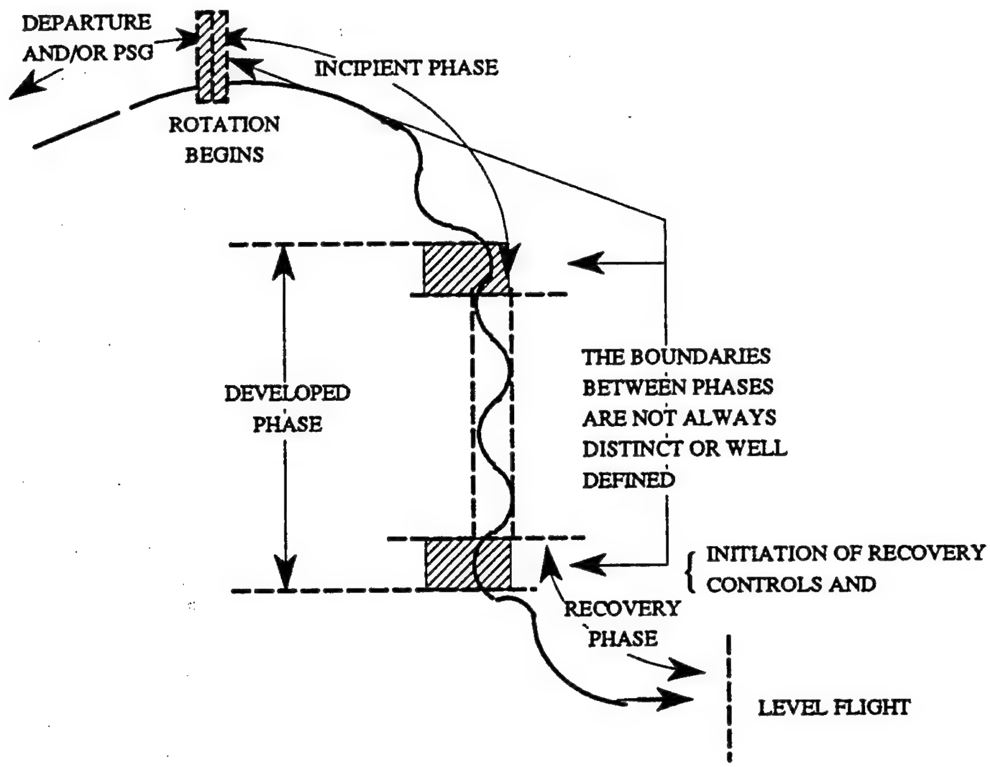


FIGURE 10.12. SPIN PHASES

10.3.5.1 INCIPIENT PHASE

The incipient phase of a spin is the initial, transitory part of the motion during which it is impossible to identify the spin mode. However, notice in Figure 10.12 that the yaw rotation begins as the incipient phase begins; that is, the visual cue to the pilot is of a sustained (though by no means steady) yaw rotation. A further distinction between the PSG (if one occurs) and the incipient phase of the spin is that the angle of attack is continuously above the stalled angle of attack (α_S) for the aircraft in the incipient phase of the spin. During a PSG the angle of attack may intermittently be less than α_S. This incipient phase continues until a recognizable spin mode develops, another boundary very difficult to establish precisely. In fact the test pilot may not recognize such a mode until he has seen it several

times; but careful examination of data traces and film may reveal that a "recognizable" mode has occurred. In this case "recognizable" does not necessarily mean recognizable in flight, but distinguishable to the engineer from all available data. In short, the incipient phase of the spin is a transitory motion easily confused with a PSG, but distinctly different from either a PSG or the developed phase of the spin.

10.3.5.2 DEVELOPED PHASE

The developed phase of a spin is that stage of the motion in which it is possible to identify the spin mode. During this phase it is common for oscillations to be present, but the mean motion is still abundantly clear. The aerodynamic forces and moments are not usually completely balanced by the corresponding linear and angular accelerations, but at least equilibrium conditions are being approached. Generally it is evident in the cockpit that the developed phase is in progress, though the exact point at which it began may be quite fuzzy. Since the aircraft motion is approaching an equilibrium state, it is frequently advisable to initiate recovery before equilibrium is achieved. For example, during the T-38 test program warning lights were installed to signal a buildup in yaw rate. Test pilots initiated recovery attempts when these lights came on. Still, in the flat spin mode with recovery initiated at 85° per second, a peak yaw rate of 165° per second was achieved. The longitudinal acceleration at the pilot's station was approximately 3.5 g and the spin was terminated by deployment of the spin chute (10.8: 10, 11). The developed spin, while it may be more comfortable due to less violent oscillations, can be deceptively dangerous, and the spin phase which follows can be disastrous.

10.3.5.3 FULLY DEVELOPED PHASE

A fully developed spin is one in which the trajectory has become vertical and no significant change in the spin characteristics is noted from turn to turn. Many aircraft never reach this phase during a spin, but when they do, they are often very difficult to recover. The smooth, flat spin of the F-4 is a classic example whereby this phase is attained and from which there is no known aerodynamic means of recovery. But a fully developed spin obviously requires time and altitude to be generated; how much time and how much altitude are strong

functions of entry conditions. As a general rule, departures that occur at high airspeeds (high kinetic energy) require more time and altitude to reach the fully developed phase than departures which occur at low kinetic energy. Finally, the spin characteristics that remain essentially unchanged in the fully developed phase include such parameters as time per turn, body axis angular velocities, altitude loss per turn, and similar quantities. However, the definition does not prohibit a cyclic variation in any of these parameters. Hence a fully developed spin can be oscillatory.

With this rather lengthy set of definitions in mind, it is now appropriate to look more closely at spinning motions and at the aerodynamic and inertial factors which cause them and the PSG.

10.3.6 THE SPINNING MOTION

Because the PSG is a random and usually a highly irregular motion, it is very difficult to study. On the other hand, the spin can approach an equilibrium condition and is therefore much more easily understood. Further, since the PSG is affected by the same aerodynamic and mass loading characteristics as the spin, an understanding of the spin and the factors affecting it are appropriate to the purposes of this course.

10.3.6.1 FLIGHTPATH DESCRIPTION

An aircraft spin is a coupled motion at extreme attitudes that requires all six equations of motion for a complete analysis. It is usually depicted with the aircraft center of gravity describing a helical path as the airplane rotates about an axis of rotation. Figure 10.13 shows such a motion. Notice that the spin axis of rotation may be curved and that the spin vector ω is constantly changing. Such a motion is highly complex, but by making some approximations a simplification results which can be very useful in understanding the spin and its causes.

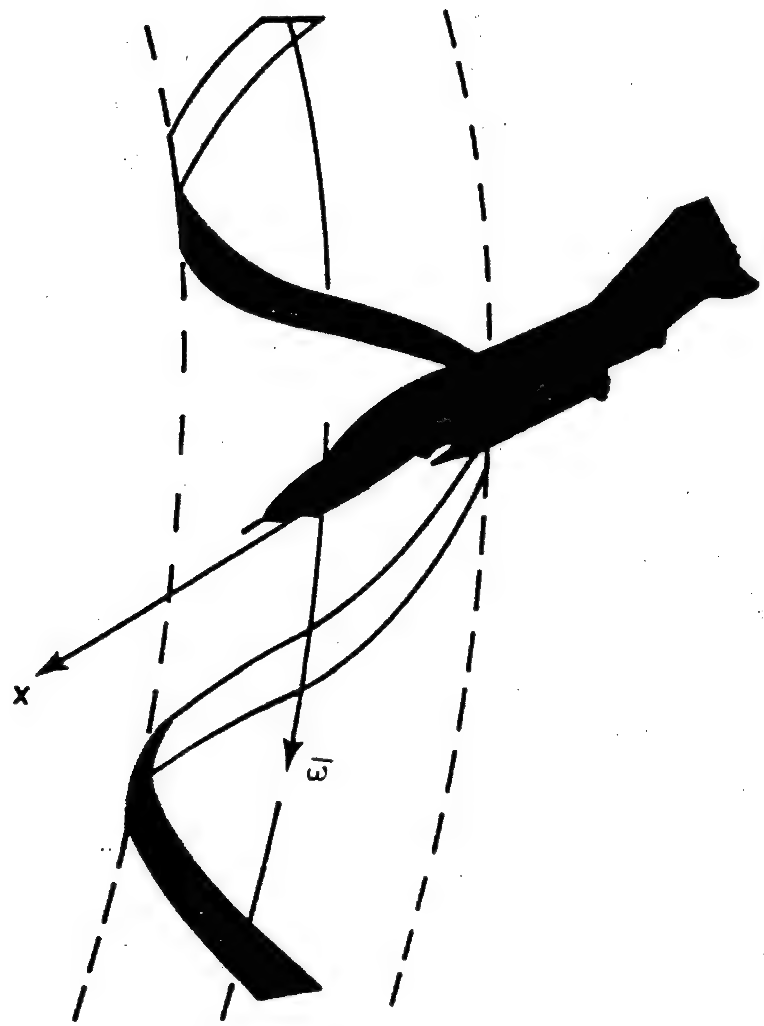


FIGURE 10.13. HELICAL SPIN MOTION

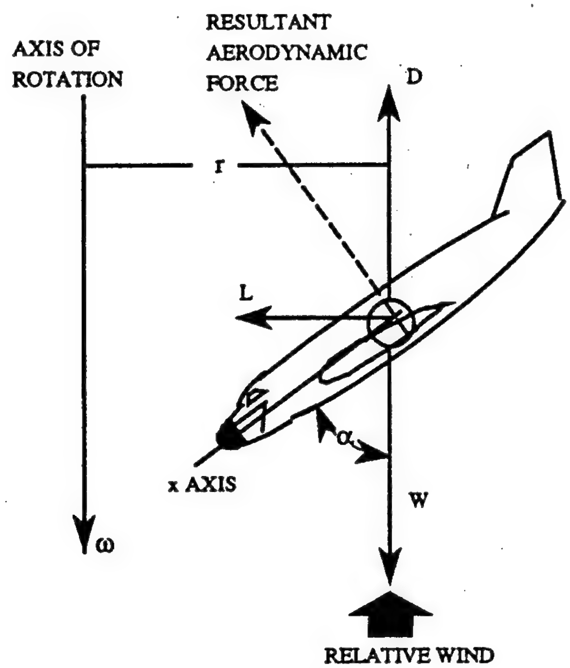


FIGURE 10.14. FORCES IN A STEADY SPIN WITHOUT SIDESLIP

In a fully developed spin with no sideslip the spin axis is vertical as indicated in Figure 10.14.

If side force is ignored, the resultant aerodynamic force acts in the x-z plane and is approximately normal to the wing chord. Taking the relative wind to be nearly vertical, a summation of vertical forces gives

$$W = D = 1/2 \ \rho V^2 S \, C_D \qquad 10.6$$

A similar summation of horizontal forces suggests that the lift component L balances the centrifugal force so that

$$mr\,\omega^2 = L = 1/2 \ \rho V^2 S \, C_L \qquad 10.7$$

Equation 10.6 suggests that as AOA increases (and C_D increases), the rate of descent (V) must decrease. Furthermore, at a stalled AOA, C_L decreases as AOA increases. With these two facts in mind it is clear that the left hand side of Equation 10.7 must decrease as the AOA

increases in a spin. The rotation rate, ω, tends to increase as AOA increases hence, the radius of turning, r, must decrease rapidly as AOA increases. These observations point up the fact that in a fully developed spin, ω and the relative wind are parallel, and become more nearly coincident as the AOA increases. In fact the inclination (η) of the flightpath (relative wind) to the vertical is given by

$$\tan \eta = \frac{r\,\omega}{V}$$

A typical variation of η with AOA is from about 5.5° at $\alpha = 50°$ to 1° at $\alpha = 80°$ (10.8:533). So, it is not farfetched to assume that ω is approximately parallel to the relative wind in a fully developed spin. All of these observations have been made under the assumption that the wings are horizontal and that sideslip is zero. The effects of bank and sideslip, while extremely important, are beyond the scope of this course, but References 10.8 and 10.9 offer some insight. It is noteworthy that this simplified analysis is valid only for a fully developed spin. However, the trends of the underlying physical phenomena will give a greater appreciation of the other phases of the spin and of the post-stall gyration.

10.3.6.2 AERODYNAMIC FACTORS

In the post-stall flight regime the aircraft is affected by very different aerodynamic forces than those acting upon it during unstalled flight. Many aerodynamic derivatives change sign; others which are insignificant at low angles of attack become extremely important. Probably the most important of these changes is a phenomenon called autorotation which stems largely from the post-stall behavior of the wing.

10.3.6.3 AUTOROTATIVE COUPLE OF THE WING

If a wing is operating at α_1 (low angle of attack) and experiences a $\Delta\alpha$ due to wing drop, there is a restoring moment from the increased lift. If, on the other hand, a wing operating at α_2 ($\alpha_2 > \alpha_s$) experiences a sudden drop, there is a loss of lift and an increase in

drag that tends to prolong the disturbance and sets up autorotation. These aerodynamic changes are illustrated in Figure 10.15.

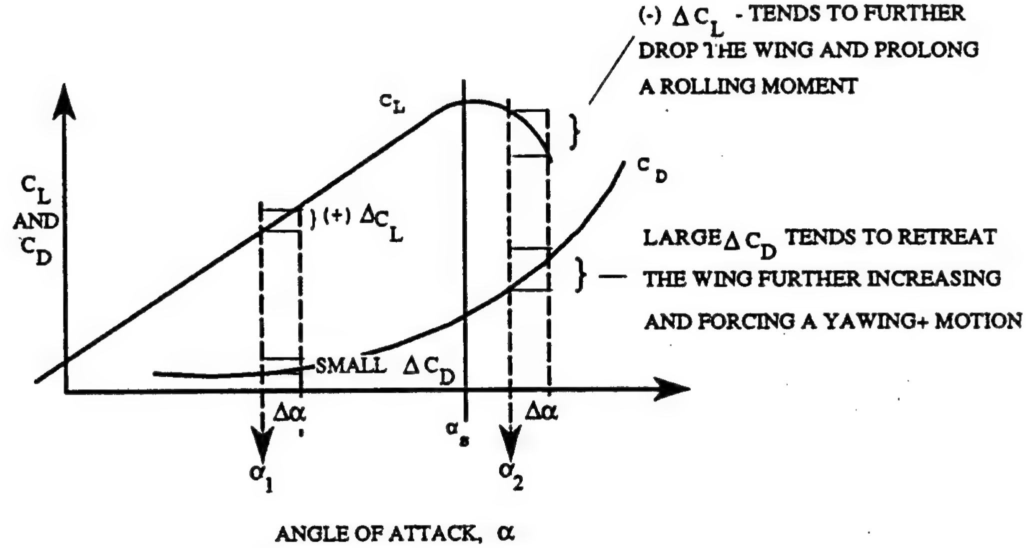


FIGURE 10.15. CHANGES IN C_L AND C_D WITH $\alpha < \alpha_s$ AND $\alpha > \alpha_s$

Consider now a wing flying in the post-stall region of Figure 10.15 and assume that some disturbance has given that wing an increase in α which tends to set up a yawing and rolling motion to the right as shown in Figure 10.16. The angle of attack of the advancing wing (Section A) corresponds to α_2 in Figure 10.15 while the angle of attack of the retreating wing (Section R) corresponds to $\alpha_2 + \Delta_\alpha$ in Figure 10.15. Figure 10.17 shows these two sections and illustrates why the advancing wing is operating at a lesser angle of attack than the retreating wing. In each case the velocity vectors are drawn as they would be seen by an observer fixed to the respective wing section. The difference in the resultant aerodynamic forces, R_A and R_R are resolved into components along the xyz body axis as in Figure 10.18. Notice that ΔF_x is in a positive x-direction, while ΔF_z is in a negative z-direction. ΔF_x forms a couple as depicted in Figure 10.19 that tends to sustain the initial yawing moment to the right. Of course, ΔF_z contributes a similar rolling couple about the x-axis which tends to sustain the initial rolling moment to the right. Ordinarily, this

autorotation generated by the wing is the most important aerodynamic factor causing and sustaining a spin. However, the other parts of the aircraft also have a part to play.

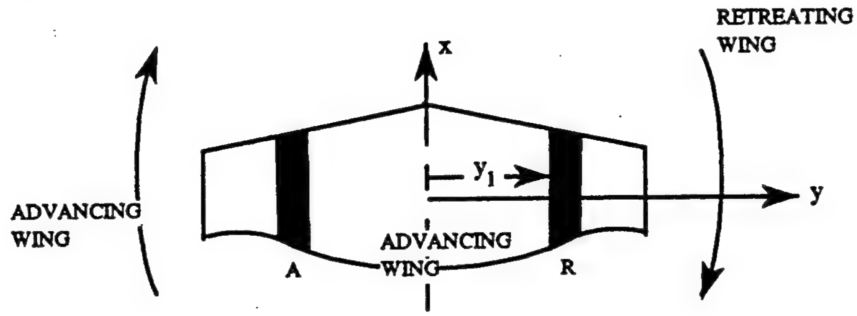


FIGURE 10.16. PLAN VIEW OF AUTOROTATING WING

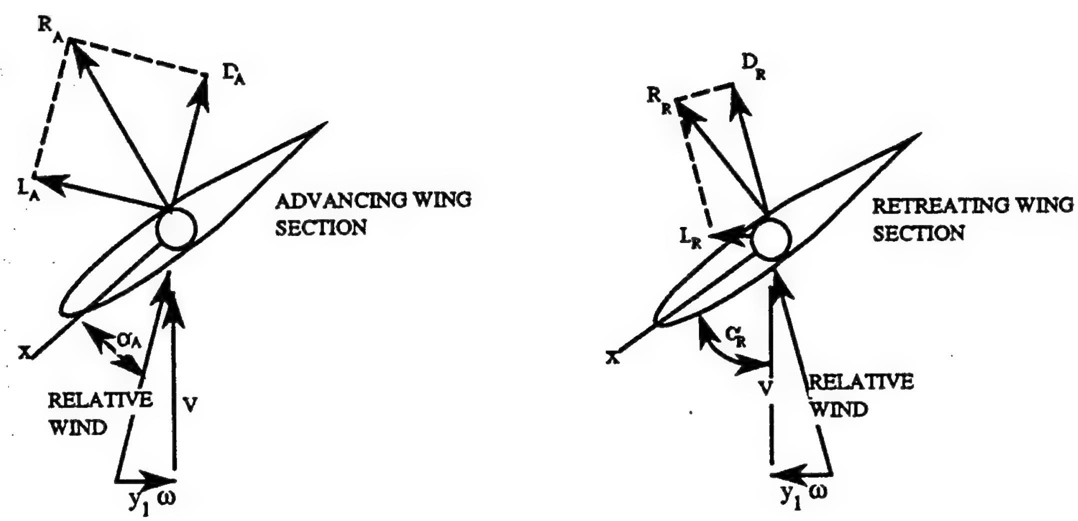


FIGURE 10.17. DIFFERENCE IN AOA FOR THE ADVANCING AND RETREATING WING IN AUTOROTATION

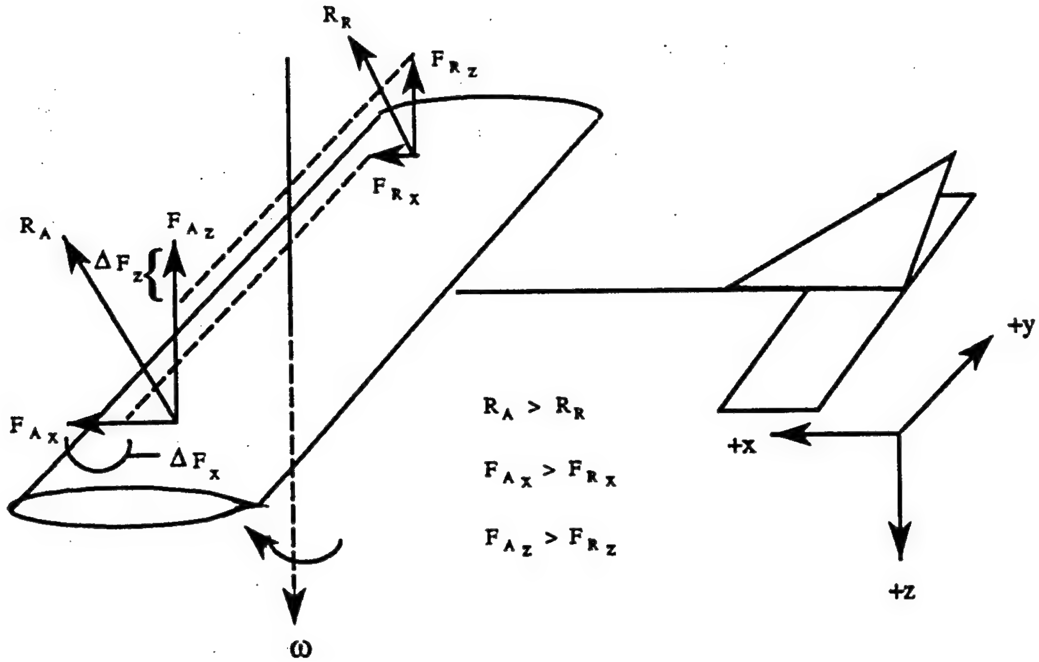
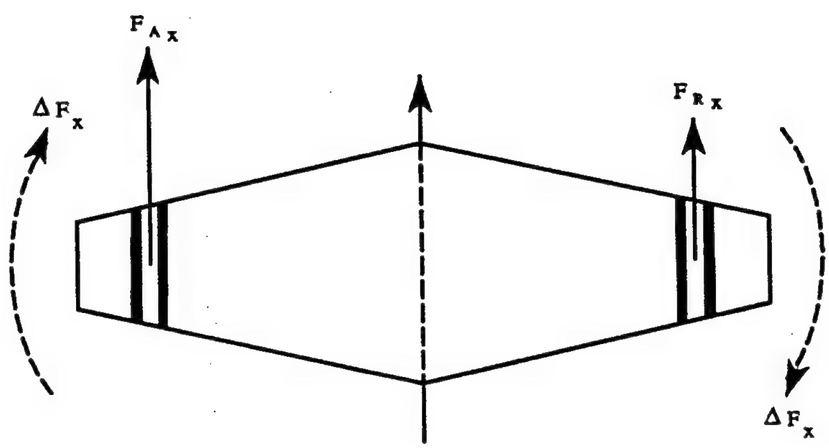


FIGURE 10.18. DIFFERENCE IN RESULTANT AERODYNAMIC FORCES



$F_{A_X} - F_{R_X} = \Delta F_X$ YAW SUSTAINING

SIMILARLY

$F_{A_Z} - F_{R_Z} = \Delta F_Z$ ROLL SUSTAINING

FIGURE 10.19. AUTOROTATIVE YAWING COUPLE

10.3.6.4 FUSELAGE CONTRIBUTIONS

The aerodynamic forces on the fuselage at stalled angles of attack are very complex, highly dependent on fuselage shape, and may either oppose or increase the autorotative couples. Sidewash flow over the fuselage greatly affects the dihedral effect $C_{l\beta}$ and may even increase it to values greater than those observed for unstalled flight (10.8:529). Weathercock stability $C_{n\beta}$ will also be affected significantly by sideshape, as illustrated in Figure 10.20. The fuselage in Figure 10.20A acts much like an airfoil section and may well generate a resultant aerodynamic force which would contribute to the yawing autorotative couple. Of course the fuselage shape will determine the relative size of "lift" and "drag" contributed by the rotating nose section. A box-like fuselage cross-section will probably give a resultant aerodynamic force opposing the yaw autorotation.

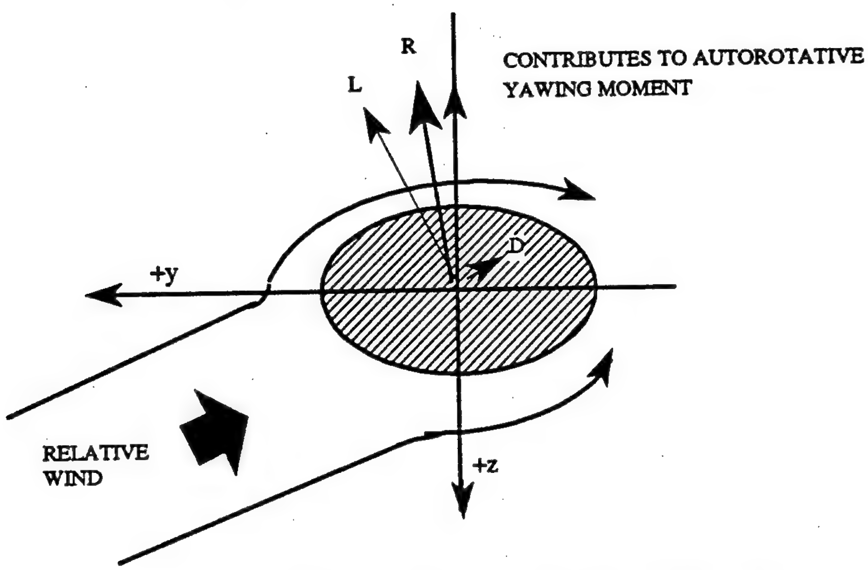


FIGURE 10.20A. PLAIN FUSELAGE

An extreme example of this type of fuselage cross-section reshaping is the strakes added to the nose of the T-37, as in Figure 10.20B. Clearly the flow separation produced by the strakes in a flow field with considerable sidewash reorients the resultant aerodynamic force in such a way as to produce an anti-spin yawing moment. Such devices have also been used on the F-111, F-16, F-18, and F-20.

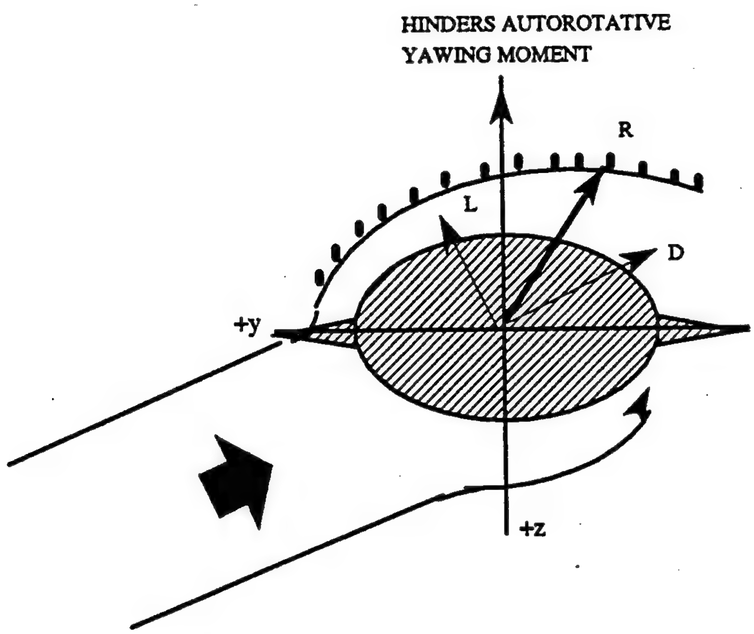


FIGURE 10.20B. FUSELAGE WITH STRAKES

10.3.6.5 CHANGES IN OTHER STABILITY DERIVATIVES

All of the other stability derivatives, especially those depending on the lift curve slope of the wing, behave in a different manner in the post-stall flight regime. However, a more complete discussion of the post-stall behavior of such derivatives as C_{l_p}, C_{n_p}, C_{n_r}, and combinations of these derivatives is given in (10.8:529). For the purposes of this course it suffices to say that C_{l_p} becomes positive and C_{n_r} may become positive in the post-stall flight regime; C_{n_r} may also become greater in stalled flight. Each of these changes contributes to autorotation, the aerodynamic phenomenon which initiates and sustains a spin. However, aerodynamic considerations are by no means the only factors affecting the post-stall motions of an aircraft. The inertia characteristics are equally important.

10.3.6.6 AIRCRAFT MASS DISTRIBUTION

10.3.6.6.1 Inertial axes. For every rigid body there exists a set of inertial axes for which the products of inertia are zero and one of the moments of inertia is the maximum possible for the body. For a symmetrical aircraft, this inertial axis system is frequently quite close to the body axis system. For the purpose of this course, the small difference in displacement is neglected, and the inertial axes are assumed to lie along the body axes. Figure 10.21 illustrates what the actual difference might be.

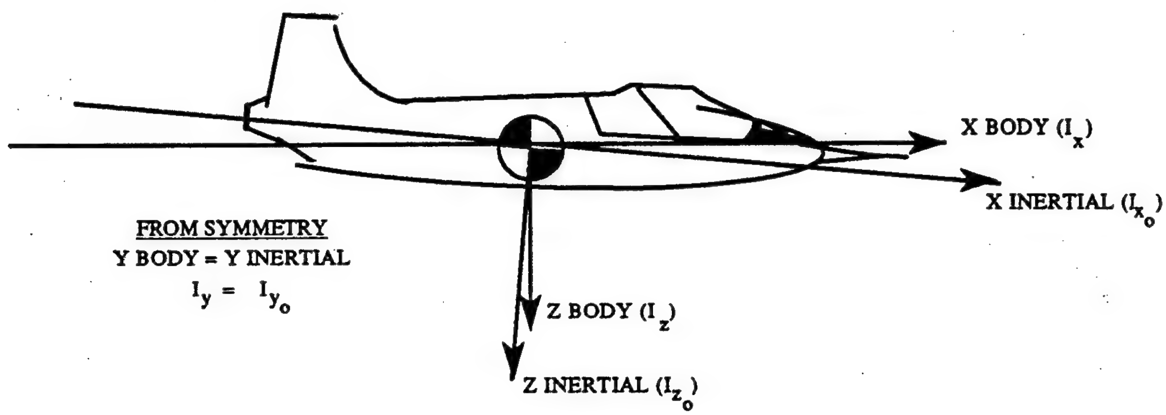


FIGURE 10.21. BODY AND INERTIAL AXES PROXIMITY

10.3.6.6.2 Radius of gyration. The center of gyration of a body with respect to an axis is a point at such a distance from the axis that, if the entire mass of the body were concentrated there, its moment of inertia would be the same as that of the body. The radius of gyration (K) of a body with respect to an axis is the distance from the center of gyration to the axis. In equation form

$$\int (y^2 + z^2)\ dm = I_x = K_x^2 m$$

$$\int (x^2 + z^2)\ dm = I_y = K_y^2 m$$

$$\int (x^2 + y^2)\ dm = I_z = K_z^2 m$$

or

$$K_i^2 = I_i/m, \tag{10.8}$$

where:

$i = x, y, \text{ or } z$

10.3.6.6.3 Relative aircraft density. A non-dimensional parameter called relative aircraft density (μ) is frequently used to compare aircraft density to air density.

$$\mu = \frac{m/Sb}{\rho} = \frac{m}{\rho Sb} \tag{10.9}$$

10.3.6.6.4 Relative magnitude of the moments of inertia. The aircraft mass distribution is frequently used to classify the aircraft according to loading. Because aircraft are "flattened" into the xy plane, I_z is invariably the maximum moment of inertia. I_x is greater or less than I_y depending on the aircraft's mass distribution. The relative magnitudes of the moments of inertia are shown in Figure 10.22. As will be seen in the next paragraph the relative magnitudes of I_x, I_y, and I_z are of utmost importance in interpreting the equations of motion.

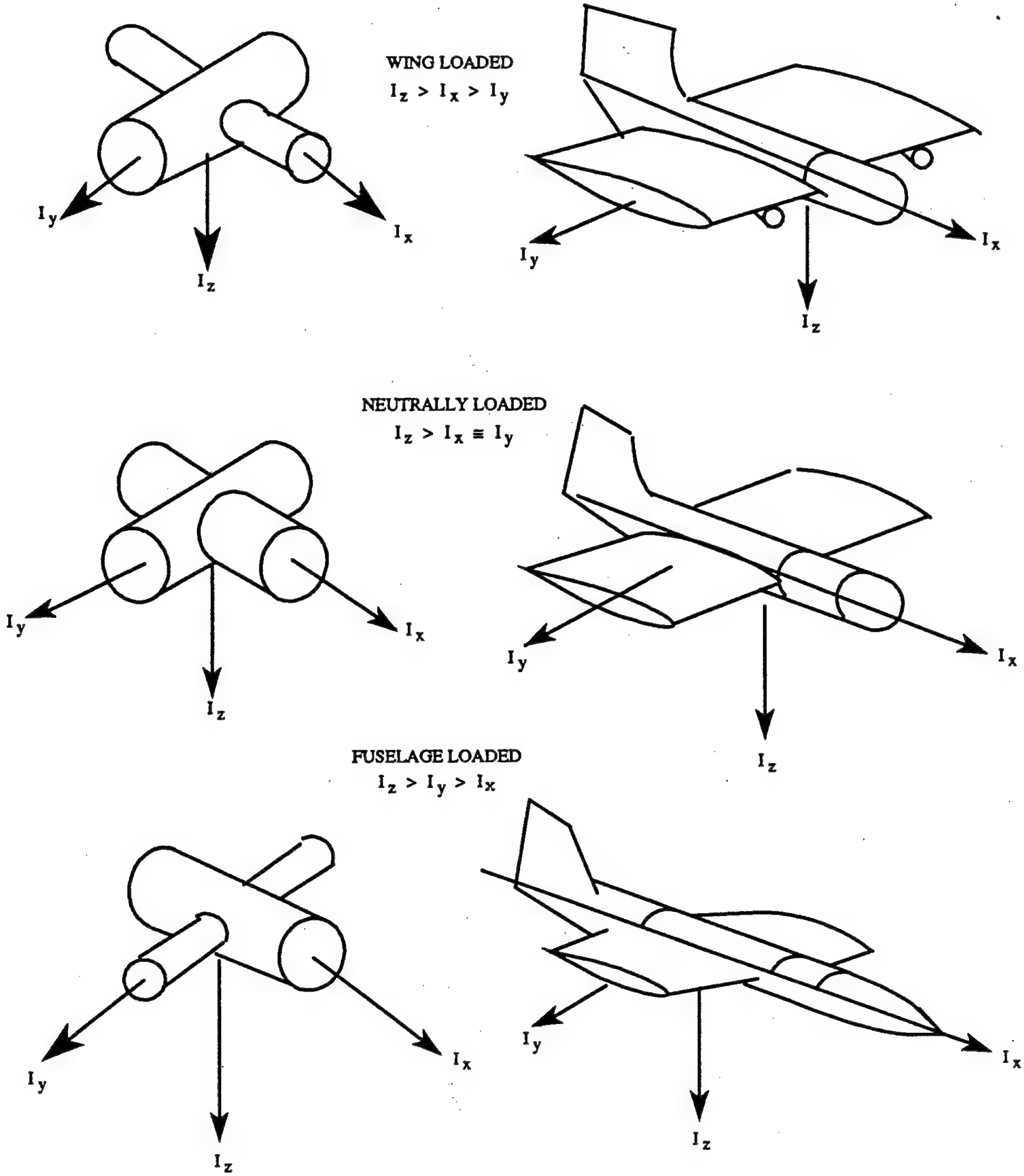


FIGURE 10.22. AIRCRAFT MASS DISTRIBUTION

10.3.7 EQUATIONS OF MOTION

Maneuvers within the post-stall flight regime can be analyzed by using all six equations of motion and integrating them numerically on a computer. From such studies, predictions of rate of rotation, angle of attack, magnitude of oscillations, optimum recovery techniques, and other parameters can be made. However, such studies must use rather inaccurate theory to predict stability derivatives or else depend on wind tunnel data or free flight model tests to provide the aerodynamic data. Hence, many researchers prefer to rely almost completely on model tests for predictions prior to flight tests. Correlation between model tests and aircraft flight tests is generally good. But model tests also have limitations. Spin tunnel tests primarily examine developed or fully developed spins; there is no good way to investigate PSGs or the incipient phase of the spin in the spin tunnel. Reynolds number effects on both spin tunnel and free flight models make it very difficult to accurately extrapolate to the full scale aircraft. Engine gyroscopic effects are not often simulated in model tests. Finally, model tests are always done for a specific aircraft configuration, which is a distinct advantage for a flight test program even though it does not suit the purposes of this course. However, it would be foolish to ignore either computer analyses or model test in preparing for a series of post-stall flight tests. For obvious reasons, this course will be restricted to a much simplified look at the equations of motion as applied to a fully developed spin.

10.3.7.1 ASSUMPTIONS

The analytical treatment used in this chapter is based on many simplifying assumptions, but even with these assumptions, good qualitative information can be obtained. The most important assumption is that only a fully developed spin with the wings horizontal will be considered. The wings horizontal, fully developed spin involves a balance between applied and inertial forces and moments. Some of the ramifications of this assumption are:

a. Initially, it will also be assumed that the applied moments consist entirely of aerodynamic ones, although other factors will be considered in later paragraphs.

b. With the wings horizontal, ω lies entirely within the xz plane. Also, with the aerodynamic and inertia forces balanced, $q = 0$, ie $\omega = pi + rk$.

c. The rate of descent (V) is virtually constant, as is altitude loss per turn.

d. V and ω are parallel.

e. The time per turn is constant, or ω is constant. Hence, $\dot{p} = \dot{q} = \dot{r} = 0$.

10.3.7.2 GOVERNING EQUATIONS

The reference frame for expressing moments, forces, accelerations, etc., is the xyz body axis frame which rotates at the same rate as the spin rotation rate ω. The origin of the xyz axes is centered at the aircraft's cg and translates downward at a rate equal to the constant rate of descent V. With this background the forces acting on the aircraft can be examined.

10.3.7.2.1 Forces. The external forces applied to the aircraft and expressed in an inertial reference frame follow Newton's second law.

$$F = m\dot{V}$$

Expressing V in the xyz reference frame,

$$F = m(\dot{V} + \omega V)$$

but since V is constant in the fully developed spin and since ω and V are parallel,

$$F = 0$$

The elimination of the force equations in this fashion merely reinforces the idea that the rotary motion is the important motion in a spin and one would expect the significant equations to be the moment equations.

10.3.7.2.2 Moments. The moment equations to be considered have already been developed in Chapter 4 and are repeated below.

$$G_x = \dot{p} I_x + qr (I_z - I_y) - (\dot{r} + pq) I_{xz} \qquad 4.5$$

$$G_y = \dot{q} I_y - pr (I_z - I_x) + (p^2 - r^2) I_{xz} \qquad 4.3$$

$$G_z = \dot{r} I_z + pq (I_y - I_x) + (qr - \dot{p}) I_{xz} \qquad 4.6$$

Using the assumption that the body axes, xyz, are also the inertial axes and considering G to consist of aerodynamic moments only, these equations become

$$L_A = \dot{p} I_x + qr (I_z - I_y) \quad \text{ROLLING MOMENT} \qquad 10.10$$

$$M_A = \dot{q} I_y - pr (I_z - I_x) \quad \text{PITCHING MOMENT} \qquad 10.11$$

$$N_A = \dot{r} I_z + pq (I_y - I_x) \quad \text{YAWING MOMENT} \qquad 10.12$$

WHERE: L_A, M_A and N_A represent the aerodynamic moments in the x, y and z body axes, respectively.

Solving for the angular accelerations shows the contributions of each type of moment to that acceleration.

$$\dot{p} = \frac{L_A}{I_x} + \frac{I_y - I_z}{I_x} qr \qquad 10.13$$

$$\dot{q} = \frac{M_A}{I_y} + \frac{I_z - I_x}{I_y} pr \qquad 10.14$$

$$\dot{r} = \frac{N_A}{I_z} + \frac{I_x - I_y}{I_z} pq \qquad 10.15$$

aerodynamic term   inertial term

The body axis angular accelerations can also be exercised in terms of aerodynamic coefficients. and the relative aircraft density.

$$\frac{L_A}{I_x} = \frac{\frac{1}{2}\rho V^2 Sb}{K_x^2 m} C_l = \frac{V^2}{\frac{2m}{\rho Sb} K_x^2} C_l$$

or

$$\frac{L_A}{I_x} = \frac{V^2}{2\mu K_x^2} C_l \qquad 10.16$$

In a similar manner,

$$\frac{N_A}{I_z} = \frac{V^2}{2\mu K_z^2} C_n \qquad 10.17$$

It is common practice in post-stall/spin literature to define C_m on the basis of wingspan instead of on the basis of wing chord as is done in most other stability and control work. This change is made to allow a consistent definition of μ :

where: $$\mu = \frac{m}{\rho Sb} \qquad 10.18$$

and is indicated by a second subscript; that is, C_m becomes $C_{m,b}$.

Then

$$\frac{M_A}{I_y} = \frac{V^2}{2\,\mu\,K_y{}^2}\,C_{m,b} \qquad 10.19$$

Equations 10.13 through 10.15 then become

$$\dot{p} = \frac{V^2 C_l}{2\,\mu K_x{}^2} + \frac{I_y - I_z}{I_x}\,q\,r \qquad 10.20$$

$$\dot{q} = \frac{V^2 C_{m,b}}{2\,\mu K_y{}^2} + \frac{I_z - I_x}{I_y}\,pr \qquad 10.21$$

$$\dot{r} = \frac{V^2 C_n}{2\,\mu K_z{}^2} + \frac{I_x - I_y}{I_z}\,pq \qquad 10.22$$

With this brief mathematical background it is now appropriate to consider the aerodynamic prerequisites for a fully developed spin to occur.

10.3.7.3 AERODYNAMIC PREREQUISITES

For a fully developed upright spin with the wings horizontal, $\dot{p} = \dot{q} = \dot{r} = q = 0$ and Equations 10.20, 10.21, and 10.22 yield

$$C_l = 0 \qquad 10.23$$

$$\frac{-V^2\, C_{m,b}}{2\,\mu K_y{}^2} = \frac{I_z - I_x}{I_y}\,pr \qquad 10.24$$

$$C_n = 0 \qquad 10.25$$

The student should consider what these results imply about a stable condition like the fully developed spin.

10.3.7.4 PITCHING MOMENT BALANCE

By examining Equation 10.24 in association with $C_{m,b}$ versus α curve for an aircraft, it is at least possible to identify regions where a fully developed spin can occur.

First, the angle of attack must be above the stall angle of attack. This condition is obvious, since the definition of a spin demands $\alpha > \alpha_s$.

Second, $C_{m,b}$ must be opposite in sign to the inertial term on the right hand side of Equation 10.24. For an upright spin this requirement means that $C_{m,b}$ must be negative. This fact is clear if one observes that $I_z > I_x$ and that p and r are of the same sign in an upright spin (Figure 10.23).

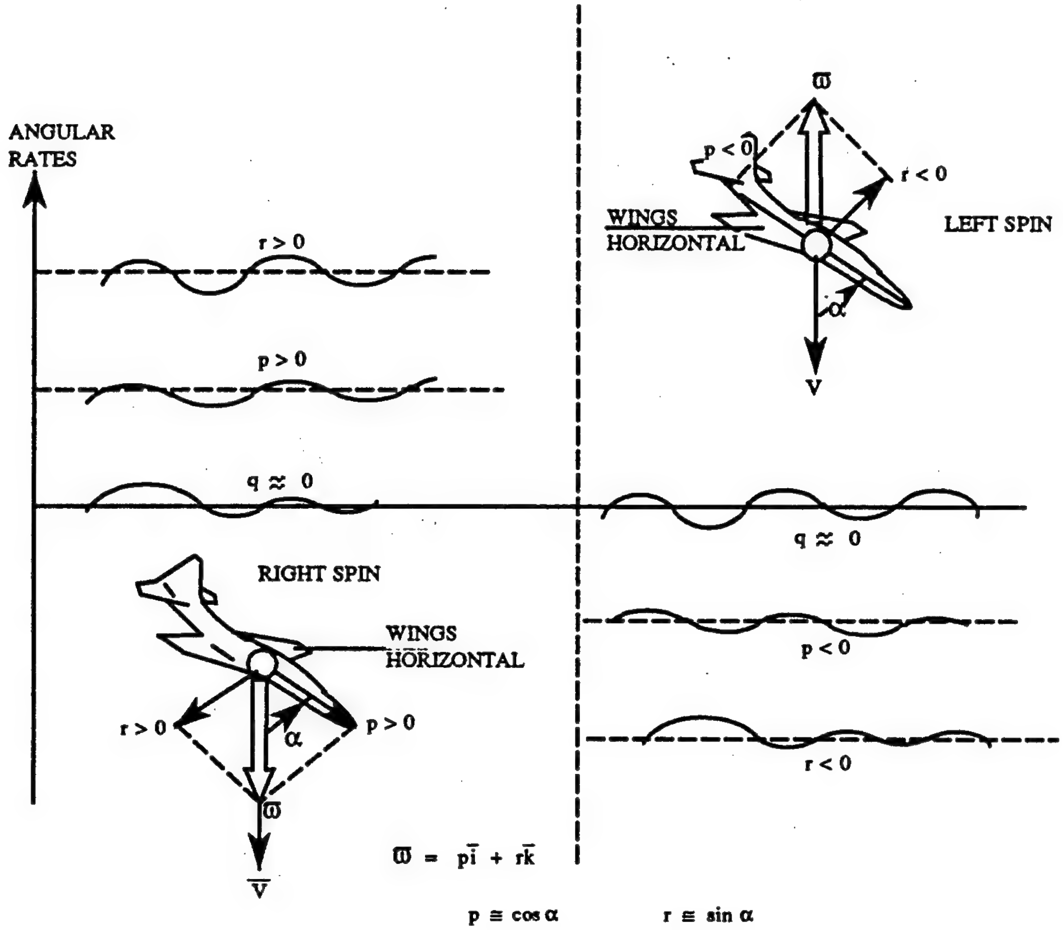


FIGURE 10.23. SPIN VECTOR COMPONENTS

In fact, it is possible to express the rotation rate in a convenient form by slightly rearranging Equation 10.24. Recall that

$$\frac{M_A}{I_y} = \frac{V^2 C_{m,b}}{2 \mu K_y^2} \qquad 10.19$$

Figure 10.23 illustrates the fact that with wings level

$$p = \omega \cos \alpha \qquad \text{and} \qquad r = \omega \sin \alpha$$

$$-\frac{M_A}{I_y} = \frac{I_z - I_x}{I_y} \omega^2 \cos \alpha \sin \alpha$$

$$\omega^2 = \frac{-M_A}{(1/2)(I_z - I_x)\sin(2\alpha)} \qquad 10.26$$

Equation 10.26 suggests that the minimum rotation rate occurs near an α of 45°, although strong variations in M_A may preclude this minimum. In fact, there is one additional prerequisite which must be satisfied before a fully developed spin can occur.

The slope of $C_{m,b}$ versus α must be negative or stabilizing and must be relatively constant. This is required simply because a positive $dC_{m,b}/d\alpha$ represents a divergent situation and would therefore require a pitching acceleration, $\dot{q} \neq 0$. But this angular acceleration would violate the assumption of a constant ω in a fully developed spin. Said another way, any disturbance in angle of attack would produce a $\Delta C_{m,b}$ tending to restore

$C_{m,b}$ to its initial value only so long as $dC_{m,b}/d\alpha < 0$. To summarize these constraints, consider Figure 10.24.

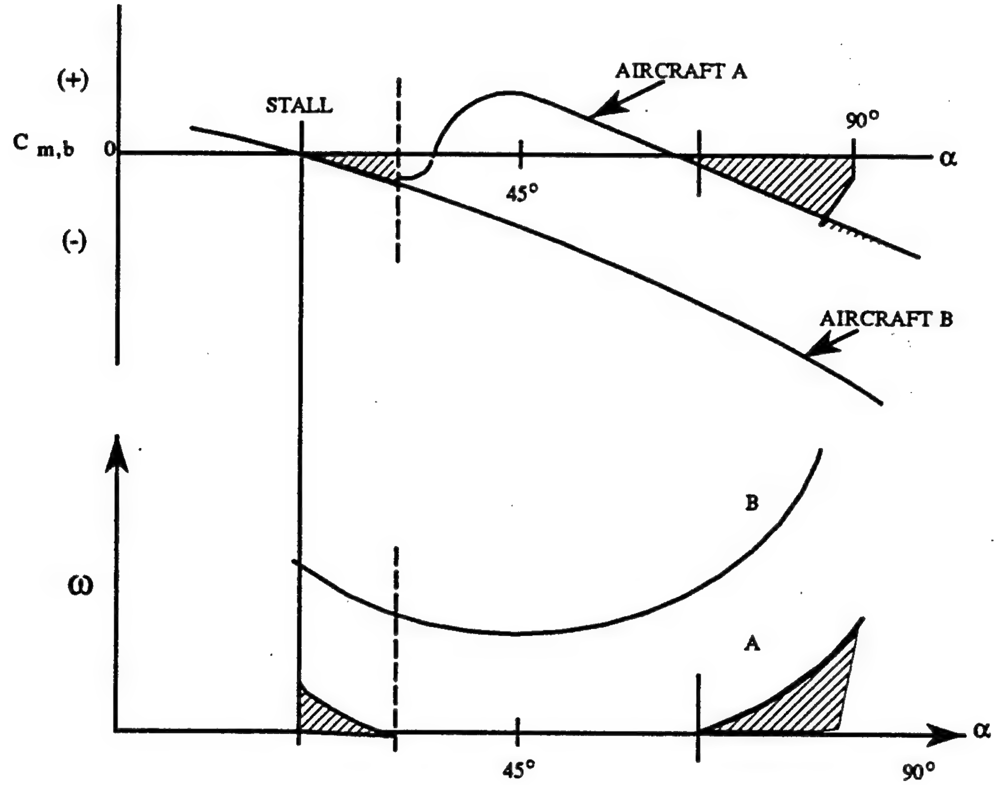


FIGURE 10.24. AERODYNAMIC PITCHING MOMENT PREREQUISITES

Aircraft B can enter a fully developed, upright spin at any AOA above α_s insofar as the pitching moment equation is concerned because its $C_{m,b}$ versus α is always negative and $dC_{m,b}/d\alpha$ is always negative. However, Aircraft A can meet the three constraints imposed by the pitching equation only in the shaded areas. Of course, the pitching moment equation is not the sole criterion; the rolling and yawing moment equations must also be considered.

10.3.7.5 ROLLING AND YAWING MOMENT BALANCE

Equations 10.23 and 10.25 suggest at least four other conditions which must be satisfied to have a fully developed spin occur. Although not specifically pointed out

previously, all the aerodynamic derivatives, even $C_{m,b}$ are functions of both α, β, and the rotation rate ω (10.10:6). Having considered $C_{m,b}$ as a function of α alone, it is convenient to consider C_n and C_l as functions of ω alone. There is little justification for this choice other than the fact the lateral-directional derivatives are more directly linked to rotation rate while the longitudinal derivative is more directly linked to angle of attack. But it is well to keep in mind that all these variables do affect $C_{m,b}$, C_l, and C_n.

The conditions imposed on both C_n and C_l to allow a fully developed spin is that they must be equal to zero, and the derivatives with respect to ω must be negative. The first of these conditions is explicitly stated by Equations 10.23 and 10.25. But the second requirement ($dC_l/d\omega < 0$ and $dC_n/d\omega < 0$) stems from the fact that a fully developed spin must be a stable condition. If an increase in ω will produce an increased C_l or C_n, then any change in rotation rate will cause the autorotative moments to diverge away from the supposedly stable initial condition. Figure 10.25 illustrates this point.

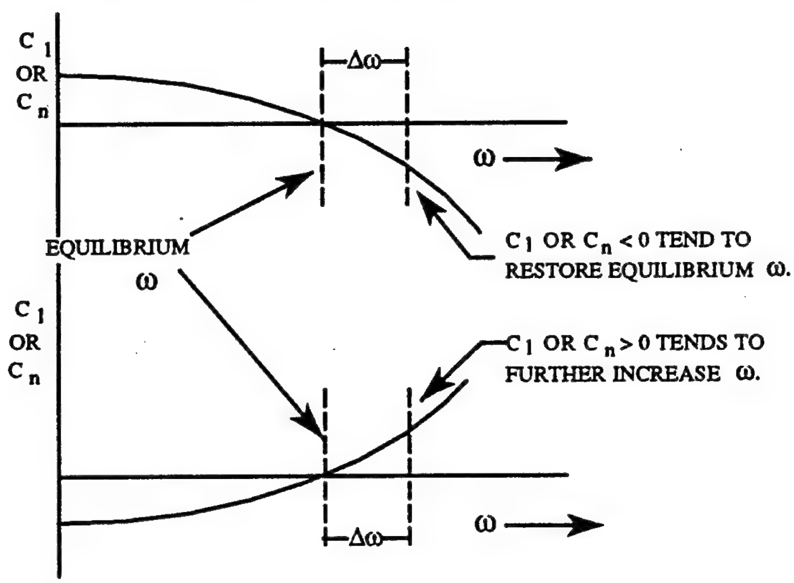


FIGURE 10.25. STABILIZING AND DESTABILIZING SLOPES FOR C_1 AND C_n VERSUS ω

Obviously, these aerodynamic prerequisites must all be met for a fully developed spin to exist in a true equilibrium form. Of course, oscillatory spins may occur with some relaxation

of one or more of these conditions. It is extremely rare to observe an ideal case which would precisely meet all these conditions in an actual spin. So, while exactly satisfying all these conditions is essential for a fully developed spin to actually exist, it is common to estimate spin parameters with less than perfect fulfillment of these prerequisites. An example of how such estimations are made will be considered next.

10.3.7.6 ESTIMATION OF SPIN CHARACTERISTICS

Reference 10.10, Appendix B, describes in detail a method of estimating spin characteristics which was designed to estimate initial conditions for a computer study investigating possible steady state spin modes of the McDonnell F-3H Demon. Although this estimation method was only intended to help predict initial conditions for the numerical integration and thus save computer time, it serves as an excellent example of how model data and the aerodynamic prerequisites discussed earlier can be combined to get a "first cut" at spin characteristics.

The aerodynamic data on which this example is based were measured by steadily rotating a model about an axis parallel to the relative wind in a wind tunnel. Hence, no oscillations in angular rates are taken into account. This limitation on the aerodynamic data is indicated by the subscribt "rb" (rotation-balance tunnel measurements). In addition, the data are presented as a function of a non-dimensional rotation rate, $\omega b/2V$. To help simplify the estimation process and partly because the rolling moment data were not as "well-behaved" as the yawing moment data, the rolling moment data were ignored. However, all the other prerequisites were observed. The estimation method is outlined below and the interested student is referred to Reference 10.10, Page 18, for a more complete description and a numerical example.

10.3.7.6.1 Determining $C_{m,rb}$ from aerodynamic data. Use the $\omega b/2V$ and α for which $C_{n,rb} = 0$ and $d\,C_{n,rb}/d\,(\omega b/2V) < 0$ to determine $C_{m,rb}$. This amounts to using the model data to determine aerodynamic pitching moments for which the aerodynamic yawing moment is zero.

10.3.7.6.2 Calculating inertial pitching moment. Using a modified form of Equation 10.26, and recognizing that the inertial pitching moment is the negative of the aerodynamic pitching moment on a fully developed spin, $-C_{m,rb}$ is calculated.

$$\omega^2 = \frac{-M_A}{(1/2)\ (I_z - I_x)\ \sin\ (2\ \alpha)} \qquad 10.26$$

$$\omega^2 = \frac{-C_{m,rb}(1/2)\rho V^2 Sb}{(1/2)\ (I_z - I_x)\ \sin\ (2\ \alpha)}$$

Solving for $-C_{m,rb}$

$$-C_{m,rb} = \frac{I_z - I_x}{\rho\ Sb} \left(\frac{\omega}{V}\right)^2 \sin(2\ \alpha) \qquad 10.27$$

10.3.7.6.3 Comparing Aerodynamic Pitching Moments and Inertial Pitching Moments. Plotting $C_{m,rb}$ versus α from the wind tunnel data and the results of Equation 10.27 on the same plot, like Figure 10.26.

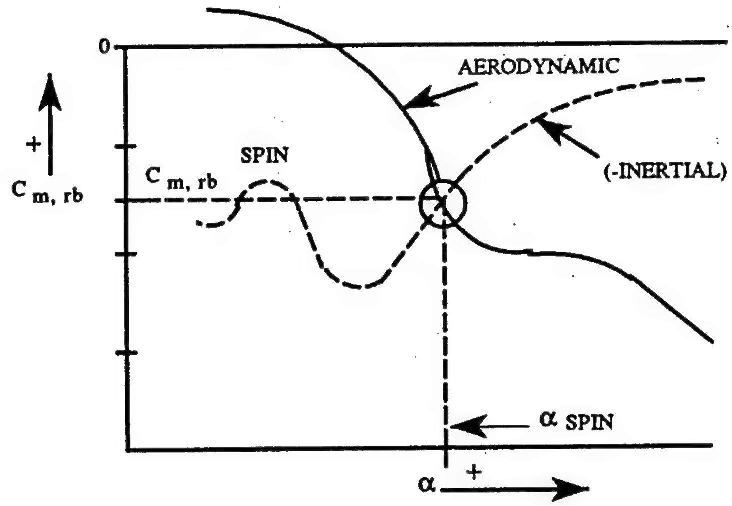


FIGURE 10.26. AERODYNAMIC PITCHING MOMENTS COMPARED TO INERTIAL PITCHING MOMENTS

The intersection of the two curves indicates a possible fully developed spin. From this plot the angle of attack of the potential spin is read directly and the value of $C_{m,rb}$ is used to calculate the potential rotation rate.

10.3.7.6.4 Calculation of ω. Rearranging Equation 10.27

$$\left(\frac{\omega}{V}\right)^2 = \frac{\left(-C_{m,rb}\right)\rho S b}{\left(I_z - I_x\right)\sin(2\alpha)} \qquad 10.28$$

the ratio ω/V can be calculated. But Equation 10.6 allows calculation of V if C_D is known. The model force measurements provide C_D and then

$$V^2 = \frac{W}{(1/2)\,\rho\, S\, C_D}$$

It follows that

$$\omega^2 = \frac{\left(-C_{m,rb}\right)\left(\rho S b\right) W}{\left(I_z - I_x\right)\left(\sin 2\alpha\right)\left(1/2\right)\rho S C_D}$$

or,

$$\omega^2 = \frac{-2\, C_{m,rb}\, b\, W}{C_D\, (I_z - I_x)\, \sin(2\alpha)} \qquad 10.29$$

10.3.7.6.5 Results. A typical set of results from the numerical integration of the six equations compared with the estimated parameters is given in Table 10.5 (10.10:26,27).

TABLE 10.5. TYPICAL COMPUTER RESULTS VERSUS ESTIMATION

Computer Results			Estimation		
α (deg)	ω (rad/sec)	V (ft/sec)	α (deg)	ω (rad/sec)	V (ft/sec)
36.0	1.88	294	38.2	1.90	285
37.0	1.92	372	45.1	1.83	327
Oscillated out of spin			48.2	1.89	453
51.8	2.18	619	50.5	2.18	620
80.0	4.72	494	70.0	3.50	515
36.5	2.80	380	37.4	2.69	365

10.3.7.6.6 Gyroscopic influences. Only aerodynamic moments have been considered so far in expanding the applied external moments. Ordinarily the aerodynamic moments are the dominant ones, but gyroscopic influences of rotating masses can also be important. The NF-104, for example, had virtually no aerodynamic moments at the top of its rocket-powered zoom profile. There is convincing evidence that gyroscopic moments from the engine dominate the equations of motion at these extreme altitudes (10.11:13). The externally applied moments should be generalized to include gyroscopic influences and other miscellaneous terms (anti-spin rockets, anti-spin chutes, etc.). The applied external moments become

$$G_x = L_A + L_{gyro} + L_{other} \qquad 10.30$$

$$G_y = M_A + M_{gyro} + M_{other} \qquad 10.31$$

$$G_z = N_A + N_{gyro} + N_{other} \qquad 10.32$$

In the next paragraph a simplified expansion of the gyroscopic terms is considered.

10.3.7.6.6.1 Gyroscopic Theory. By virtue of its rotation, a gyroscope tends to maintain its spin axis aligned with respect to inertial space. That is, unless an external torque is applied, the gyro spin axis will remain stationary with respect to the fixed stars. If a torque is applied about an axis that is perpendicular to the spin axis, the rotor turns about a third axis that is orthogonal to the other two axes. On removing this torque the rotation (precession) ceases - unlike an ordinary wheel on an axle which keeps on rotating after the torque impulse is removed.

These phenomena, all somewhat surprising when first encountered, are consequences of Newton's laws of motion. The precessional behavior represents obedience of the gyro to Newton's second law expressed in rotational form, which states that torque is equal to the time rate of change of angular momentum.

$$T = \frac{dH}{dt} \qquad 10.33$$

where T = external torque applied to the gyroscope

H = angular momentum of the rotating mass

$H = I\ \Omega$

with I = moment of inertia of the rotating mass

Ω = angular velocity of the rotating mass.

Equation 10.33 applies, like all Newton's laws, only in an inertial frame of reference. If it's assumed that H is to be expressed within a frame of reference rotating at the precession rate of the gyroscope, $H_{inertial} = H_{rotating} + \omega_p\ H$. If the gyro spin rate is unchanged then H measured in the rotating frame will be zero and Equation 10.33 becomes

$$\bar{T} = \bar{\omega}_p\ \bar{H} \qquad 10.34$$

The direction of precession for a gyro when a torque is applied is given by Equation 10.34 This direction is such that the gyro spin axis tends to align itself with the total angular momentum vector, which in this case is the vector sum of the angular moment due to the spinning rotor and the angular momentum change due the applied torque, ΔH as shown in Figure 10.27. The law of precession is a reversible one. Just as a torque input results in an angular velocity output (precession), an angular velocity input results in a torque output along the corresponding axis.

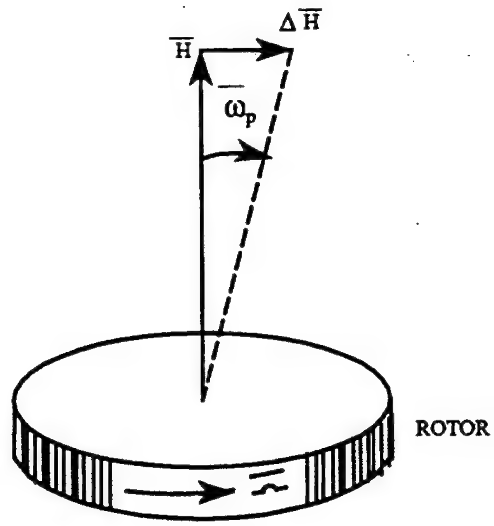


FIGURE 10.27. DIRECTION OF PRECESSION

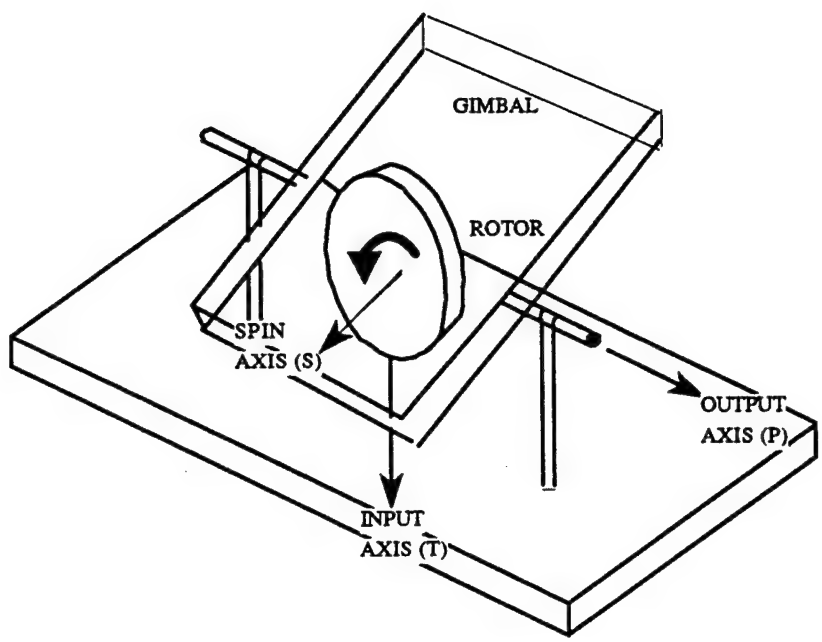


FIGURE 10.28. GYROSCOPE AXES

Three axes are significant in describing gyroscope operation; the torque axis, the spin axis and the precession axis. These are commonly referred to as input (torque), spin, and output (precession). The directions of these axes are shown in Figure 10.28. They are such

that the spin axis rotated into the input axis gives the output axis direction by the right hand rule. The direction of rotational vectors such as spin, torque, and precession can be shown by pointing the index finger of the right hand in the direction of rotation, the thumb extended will point along the axis of rotation. For gyro work, it is convenient to let the thumb, forefinger, and middle finger represent the spin, torque, and precession axes respectively. Figure 10.29 illustrates this handy memory device.

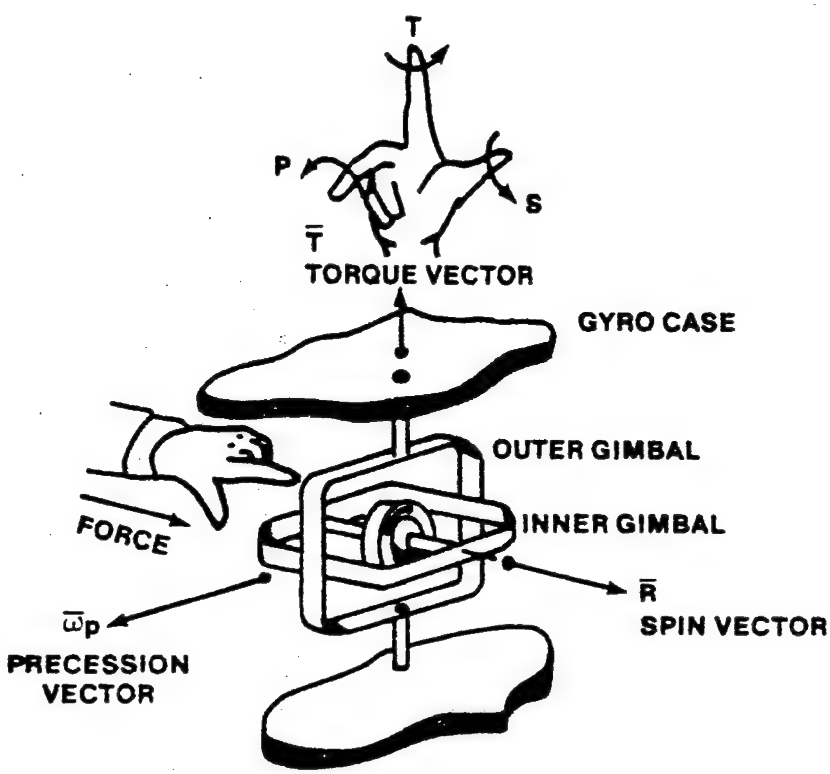


FIGURE 10.29. SPIN, TORQUE AND PRECESSION VECTORS

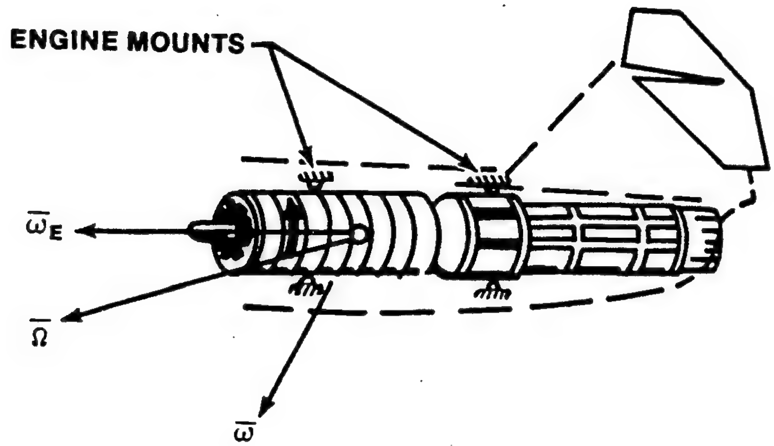


FIGURE 10.30. ANGULAR VELOCITIES OF THE ENGINE'S ROTATING MASS

10.3.7.6.6.2 Engine Gyroscopic Moments. In Figure 10.30, consider the rotating mass of the engine as a gyroscope and analyze the external torque applied to the engine mounts of an aircraft in a spin. Then the total angular velocity of the rotating mass is the vector sum of $\omega_E + \omega$, with ω_E being the engine RPM (assumed constant) and ω being the aircraft's spin rotation rate.

$$\Omega = \omega_E + \omega$$

But,

$$\omega << \omega_E$$

Therefore we can make the approximation

$$\Omega \cong \omega_E$$

If one also assumes that the rotational axis of the engine is parallel to the x-axis,

$$\Omega \equiv \omega_E i$$

Then the angular momentum of the engine is

$$H_E = I_E \, \omega_E \, i \qquad 10.35$$

with I_E = moment of inertia of the engine about the x-axis.

Considering Figure 10.30 again and applying Equation 10.34, the external torque applied to the engine must be the precession rate of the aircraft, ω, crossed into the engine's angular momentum.

$$T = \omega \; H_E \qquad 10.36$$

But the moment applied by the engine through the engine mounts to the spinning aircraft is equal but opposite in sign (Newton's Third Law).

$$G_{gyro} = - \; \omega \; H_E$$

$$L_{gyro} + M_{gyro} + N_{gyro} = \begin{vmatrix} i & j & k \\ p & q & r \\ I_{w_E} & 0 & 0 \end{vmatrix}$$

$$L_{gyro} = 0 \; i \qquad 10.37$$

$$M_{gyro} = (- I_E \, \omega_E \, r) \; j \qquad 10.38$$

$$N_{gyro} = (I_E \; \omega_E \, q) \; k \qquad 10.39$$

Then Equations 10.13, 10.14, and 10.15 can be expanded to:

AERO	INERTIAL COUPLING (sometimes called gyrodynamic term)	GYROSCOPIC TERM (an engine effect)	MISCELLANEOUS (rockets, spin chutes, etc)

$$\dot{p} = \frac{L_A}{I_x} + \frac{I_y - I_z}{I_x} q r + \frac{L_{gyro}}{I_x} + \frac{L_{other}}{I_x} \qquad 10.40$$

$$\dot{q} = \frac{M_A}{I_y} + \frac{I_z - I_x}{I_y} p r + \frac{M_{gyro}}{I_y} + \frac{M_{other}}{I_y} \qquad 10.41$$

$$\dot{r} = \frac{N_A}{I_z} + \frac{I_x - I_y}{I_z} p q + \frac{N_{gyro}}{I_z} + \frac{N_{other}}{I_z} \qquad 10.42$$

Equation 10.26 becomes:

$$\omega^2 = \frac{-M_A + I_E \omega_E r}{\left(1/2\right)\left(I_z - I_x\right) \sin\left(2\alpha\right)} \qquad 10.43$$

Equation 10.43 shows that the effect of the engine gyroscopic moment is to shift the ω vs α curves as shown in Figure 10.31. An engine that rotates in a counter clockwise direction (as viewed from the intake) will cause all aircraft to spin faster in a right, upright spin and slower in a left upright spin. Generally speaking, however, this engine gyroscopic moment is negligible in comparison to the other external moments.

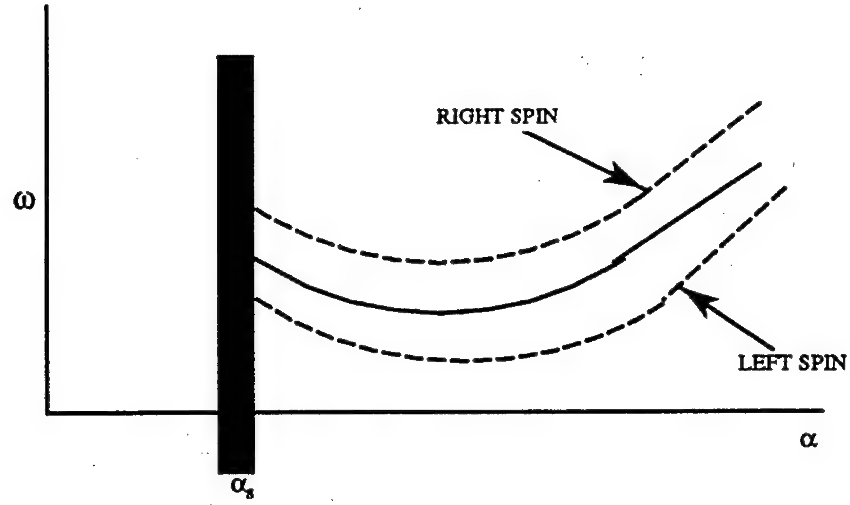


FIGURE 10.31. EFFECT OF M_{GYRO} ON SPIN ROTATION RATE

10.3.7.6.7 Spin Characteristics of Fuselage-Loaded Aircraft. It is appropriate to consider briefly some of the spin characteristics peculiar to modern high performance aircraft in which the mass is generally concentrated within the fuselage (I_y larger than I_x and almost as large as I_z). It can be shown that a system that has no external moments or forces tends to rotate about its largest principal axis, that, in the case of an aircraft, is the z axis. In an actual spinning aircraft, the external moments are not zero and thus the aircraft spins about some intermediate axis. For the idealized spin thus far considered, the pitching moment equation leads one to the observation that fuselage-loaded aircraft will probably spin flatter than their wing-loaded counterparts.

10.3.7.6.7.1 Fuselage-Loaded Aircraft Tend to Spin Flatter Than Wing-Loaded Aircraft. For a fully developed spin

$$G_y = -pr\,(I_z - I_x) \qquad 10.44$$

In an aircraft, $(I_z - I_x)$ can never be zero. Hence, if $G_y = 0$ then p must be zero, in which case $\omega = rk$ and the spin is flat ($\omega = pi$ is excluded by the definition of a spin). If the spin is not flat, then both p and r exist and, in an upright spin, have the same algebraic sign.

Because $(I_z - I_x)$ is always positive, examination of Equation 10.44 shows that G_y must always be negative (or zero) for an upright spin.

The smaller the pitch attitude (θ in Figure 10.32), the flatter the spin, and θ can be defined as $\sin^{-1}(p/\omega)$ for the spin depicted in Figure 10.32. θ varies with the relative magnitude of $(I_z - I_x)$, as can readily be seen by rearranging Equation 10.44.

$$p = \frac{\left| G_y \right|}{r\left(I_z - I_x\right)}$$

Since p becomes smaller as $(I_z - I_x)$ increases, it is clear that fuselage-loaded aircraft tend to spin flatter than wing-loaded aircraft. But what about the effect of increasing I_y upon the roll equation?

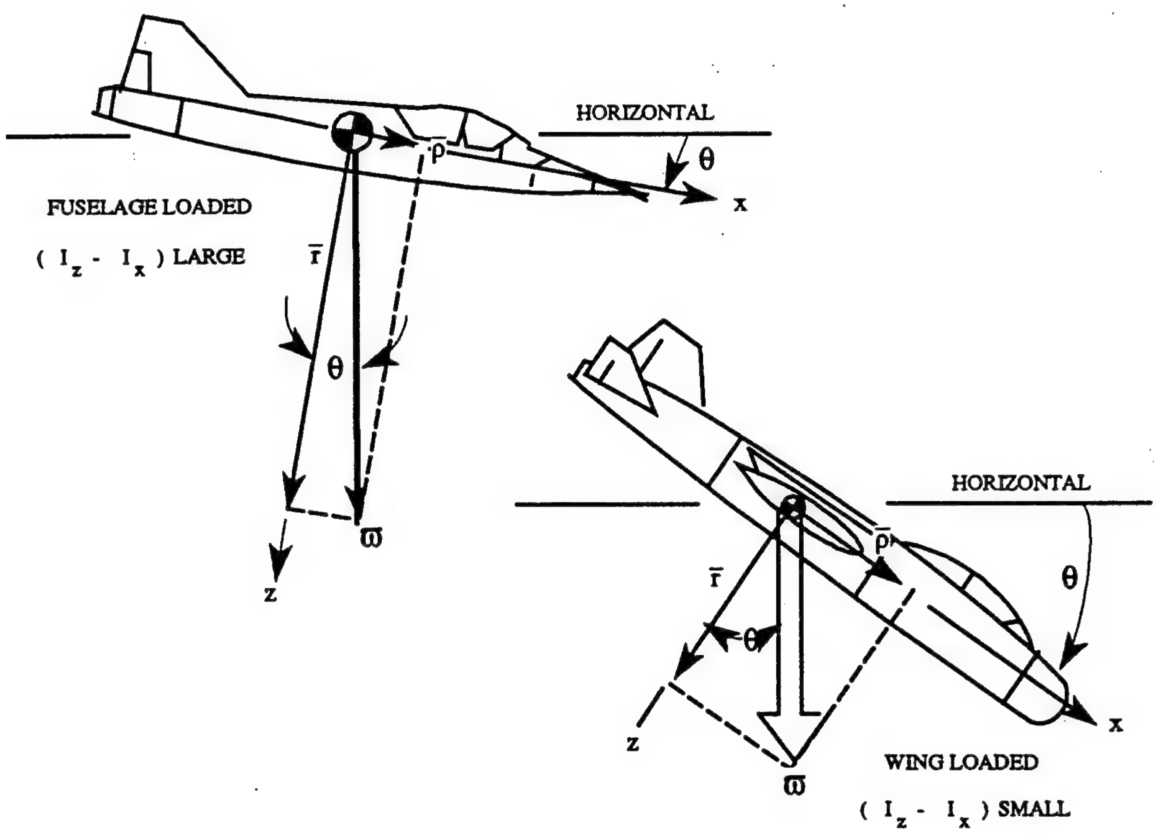


FIGURE 10.32. EFFECT OF MAGNITUDES OF I_Z AND I_X ON SPIN ATTITUDE

10.3.7.6.7.2 Fuselage-Loaded Aircraft Tend to Exhibit More Oscillations. On aircraft where I_y is approximately equal to I_z in magnitude, the fully developed spin is more likely to be oscillatory. In the limit, if $I_y = I_z$, the reference spin could be wing down, since any axis in the yz plane would be a maximum inertial axis. Although these facts suggest that the bank angle is easily disturbed and that a developed spin often occurs with the bank angle not zero, a restoring tendency does exist which leads to periodic oscillations in bank angle. Consider again the rolling moment equation.

$$G_x = \dot{p}\, I_x + q\, r\, (I_z - I_y) \qquad 10.45$$

If a "0" subscript is used to represent the reference or steady-state conditions,

$$G_{xo} = \dot{p}_o\, I_x + q_o\, r_o\, (I_z - I_y)$$

If instantaneous values are represented by Equation 10.45, the change in external moments due to the perturbations of the angular acceleration and angular velocities is

$$(G_x - G_{xo}) = (\dot{p} - \dot{p}_o)\, I_x + (q\, r - q_o\, r_o)(I_z - I_y)$$

Assuming perturbations in roll will not significantly change r_0, $r = r_0$ and

$$\Delta \dot{p} = \frac{\Delta G_x}{I_x} - \Delta q \frac{I_z - I_y}{I_x} r_o \qquad 10.46$$

The second term on the right side of Equation 10.46 serves to damp oscillations in that it reduces the ability of perturbations in rolling moment (ΔG_x) to produce perturbations in roll acceleration (Δp). For fuselage-loaded aircraft, in which (I_z - I_y) is small, the damping is much reduced. Thus, any perturbations in the motion tend to persist longer in fuselage-loaded aircraft than they do in wing-loaded aircraft.

10.3.7.7 SIDESLIP

It is beyond the scope of this course to deal with the effects of sideslip in any detail. However, it is noteworthy that sideslip need not be zero in a developed spin; in fact it usually is not. Reference 10.8, page 535, shows that sideslip in a spin arises from two sources: wing tilt with respect to the horizontal (φ) and the inclination of the flight path to the vertical (η).

$$\beta \equiv \varphi - \eta \qquad 10.47$$

If then, one considers a spin with a helical flight path as opposed to a vertical flight path, the inclination of the flight path to the vertical is positive and equal to the helix angle. Then, in order to maintain zero sideslip, the retreating wing must be inclined downwards by an amount equal to the helix angle in order to have zero sideslip. However, it is quite common

to have fully developed spins (with the spin axis vertical, not the flight path) with varying amounts of sideslip. Sideslip on a stalled wing will generally increase the lift on the wing toward which the sideslip occurs and reduce the lift on the opposite wing. It is easy to understand that a small amount of sideslip can produce a large rolling moment and thereby significantly alter the balance of rolling moments. These qualitative comments are quite cursory and the inquisitive student may wish to pursue these effects further. Reference 10.8 offers an expanded discussion, but to adequately discuss sideslip effects in any detail one must consider all three moment equations and their coupling effects. The consideration of sideslip leads to the general conclusion that the rolling couple can be balanced over a wide range of angles of attack and spin rotation rates.

10.3.7.8 INVERTED SPINS

Since PSGs are definitely uncontrolled aircraft motions, there is no guarantee that all spins will be upright. The test pilot particularly (and operational pilots as well) will continue to experience inverted spins and PSGs which may be mainly inverted aircraft motions. As Reference 10.12, page 1, points out,

"...inverted spins cannot be prevented by handbook entries that 'the airplane resists inverted spins'."

It is, therefore, essential that the test pilot have some appreciation of the nature of the inverted PSG/spin. As usual, the analytical emphasis will necessarily be restricted to the fully developed spin, but the qualitative comments which follow also apply in a general way to other types of post-stall motion.

The most common pilot reaction to an inverted post-stall maneuver is, "I have no idea what happened! The cockpit was full of surprise, dirt, and confusion." Why? First, negative g flight is disconcerting in and of itself, particularly when it is entered inadvertently. But even experienced test pilots can be upset and their powers of observation reduced in an anticipated inverted spin. This disorientation usually takes one of two forms: (1) inability to distinguish whether the motion is inverted or upright or (2) inability to determine the direction of the spin. Each of these problems will be considered separately.

10.3.7.8.1 Angle of Attack in an Inverted Spin. The angle of attack in an inverted spin is always negative (Figure 10.33). It might appear that it would be easy to determine the difference in an upright or inverted spin; if the pilot is "hanging in the straps," it is an inverted spin. Such an "analysis" is accurate in some spin modes (the Hawker Hunter has an easily recognized smooth, flat mode such as this); however, if the motion is highly oscillatory, not fully developed, or a PSG, the pilot's tactile senses are just not good enough. If the aircraft has an angle of attack indicator, this is probably the most reliable means of determining whether the maneuver is erect or inverted. Lacking an angle of attack system, the pilot must rely on the accelerometer or his sensory cues, neither of which are easy to interpret. But what about determining spin direction?

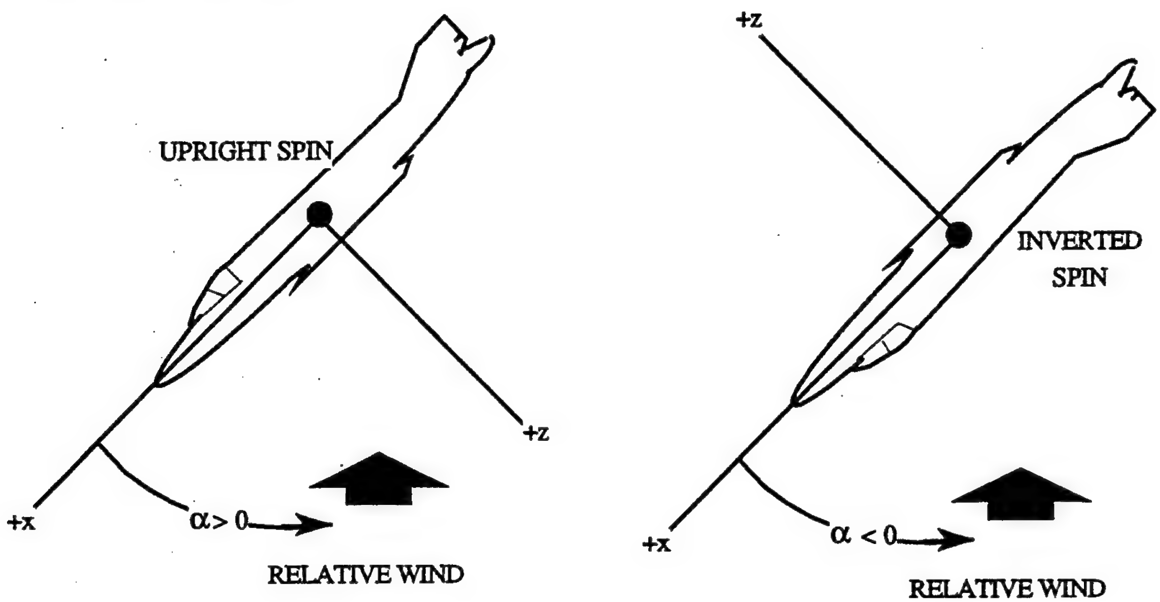


FIGURE 10.33. ANGLE OF ATTACK IN AN INVERTED SPIN

10.3.7.8.2 Roll and Yaw Directions in an Inverted Spin. Consider two identical aircraft, one in an upright spin and the other in an inverted spin as shown in Figure 10.34. Notice that the spin direction in either an upright or an inverted spin is determined by the sense of the yaw rate. Notice also that in an inverted spin the sense of the roll rate is always opposite that of the yaw rate. It is common for pilots to mistakenly take the direction of roll as the spin direction. The chances of making this error are considerably enhanced during a

PSG or the incipient phase of the spin when oscillations are extreme. In steep inverted spins ($|\alpha|$ nearly equals $|\alpha_s|$) the rolling motion is the largest rotation rate and further adds to the confusion. However, there is a reliable cockpit instrument, the turn needle, which always indicates the direction of yaw. With such confusion possible, what about the previously obtained equations of motion? Is it necessary to modify them for the inverted spin?

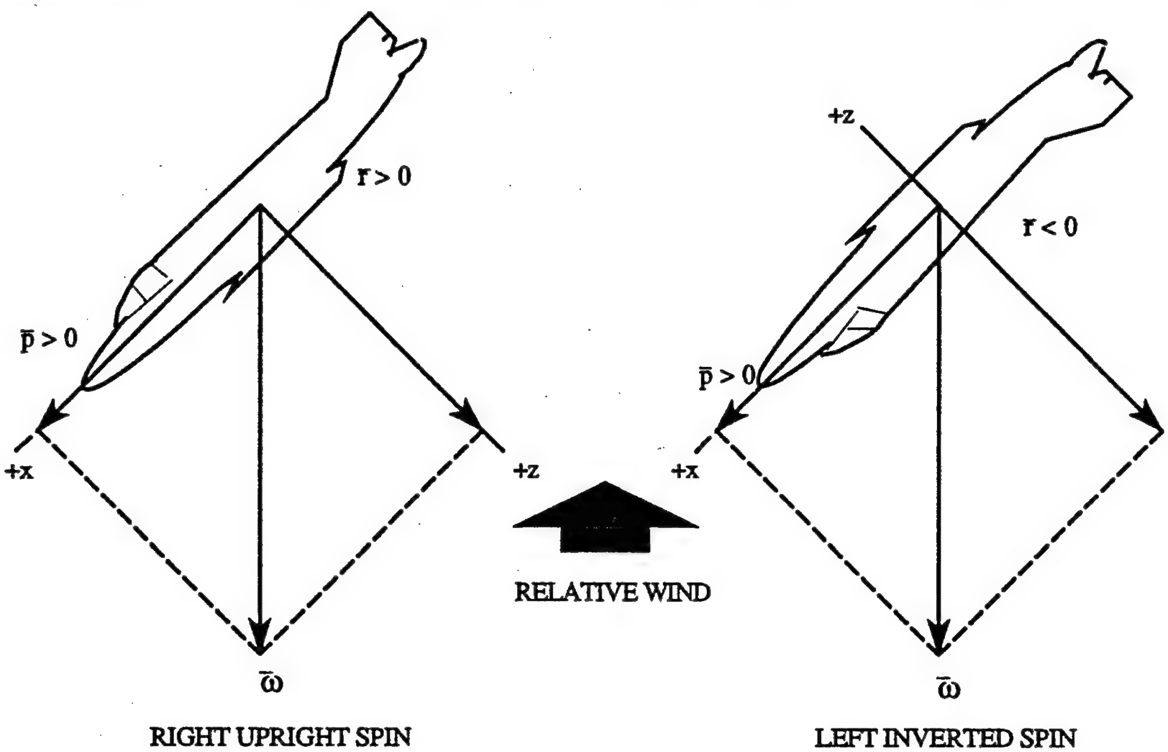


FIGURE 10.34. ROLL AND YAW RATES IN AN INVERTED SPIN

10.3.7.8.3 Applicability of Equations of Motions: All the equations previously described are directly applicable to the inverted spin. Of course, the differences in sign for angle of attack and the lack of aerodynamic data collected at negative angle of attack pose a significant practical problem in trying to do detailed analyses of the inverted spin. But for the qualitative purposes of this course, the equations of motion are usable. However, it is instructive to note the difference in the sense of the pitching moments between an upright and an inverted spin. Recall that in an upright spin, the applied external pitching moment

(dominated by the aerodynamic pitching moment) had to be negative to balance the inertial term, as Equation 10.44 for a fully developed spin shows.

$$G_y = -pr\,(I_z - I_x) \qquad 10.44$$

But when p and r are of opposite signs, as in the inverted spin, the applied external moment must be positive. This fact is illustrated in Figure 10.35, where the mass of the aircraft is represented as a rotating dumb-bell.

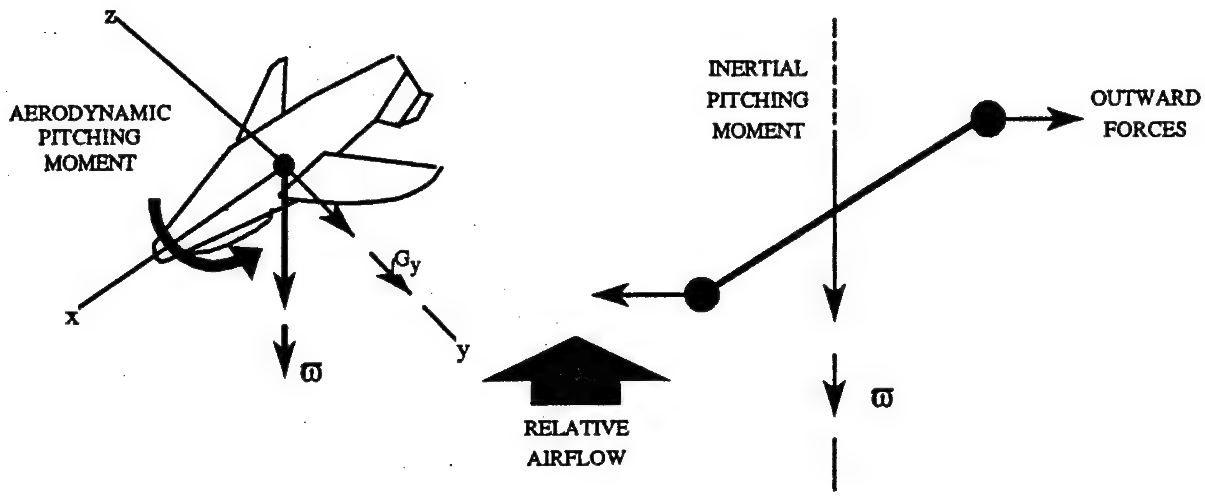


FIGURE 10.35. PITCHING MOMENTS IN AN INVERTED SPIN

It is apparent that in the inverted spin the external pitching moment is positive; that is, expressed as a vector, it lies along the positive y axis. As a final point, the recovery from PSGs/spins, both erect and inverted, must be examined in some detail.

10.3.8 RECOVERY

Obtaining developed spins today is generally difficult, but when obtained, the very factors that make this difficult may also make it difficult to recover from the spin. Current and future aircraft designs may be compromised too much for their intended uses to provide adequate aerodynamic control for termination of the developed spin; also, there is a problem of pilot disorientation associated with developed spins. As a result, the PSG and the incipient

phase of the spin must be given more attention than they have received in the past, and preventing the developed spin through good design and/or proper control techniques has become a primary consideration.

Current aircraft have greater weight and appreciably larger moments of inertia about the y and z axes than those of World War II aircraft. With the resulting high angular momentum, it is difficult for a spin to be terminated as effectively as a spin in earlier airplanes by aerodynamic controls which are generally of similar size. Furthermore, controls which are effective in normal flight may be inadequate for recovery from the spin unless sufficient consideration has been given to this problem in the design phase.

10.3.8.1 TERMINOLOGY

The recovery phase terminology was purposely omitted from previous discussion of spin phases for inclusion here. Referring to Figure 10.12, the recovery phase begins when the pilot initiates recovery controls and ends when the aircraft is in straight flight; however, there are several terms used to differentiate between the subparts of this phase.

10.3.8.1.1 Recovery. Recovery is defined as the transitional event from out-of-control conditions to controlled flight. In more useable terms, this period of time normally is counted from the time the pilot initiates recovery controls and that point at which the angle of attack is below α_s and no significant uncommanded angular motions remain. The key phrase in this expanded definition is "angle of attack below α_s;" once this objective is attained the aircraft can be brought back under control provided there are sufficient altitude and airspeed margins to maneuver out of whatever unusual attitude remains.

10.3.8.1.2 Dive Pullout and Total Recovery Altitude. The dive pullout is the transition from the termination of recovery to level flight. Total recovery altitude is the sum of the altitude losses during the recovery and dive pullout.

10.3.8.2 ALTERATION OF AERODYNAMIC MOMENTS

The balanced condition of the developed spin must be disturbed in order to effect a recovery, and prolonged angular accelerations in the proper direction are needed. Several methods for obtaining these accelerations are available but not all are predictable. Also, the accompanying effects of some methods are adverse or potentially hazardous. The general methods available for generating anti-spin moments follow with the applicable terms of the general equations also given.

Term 1 Term 2 Term 3 Term 4

$$\dot{p} = \frac{V^2}{2\mu K_x^2}\left(C_l\right) + \frac{I_y - I_z}{I_x} q\, r + 0 + \frac{L_{other}}{I_x}$$

$$\dot{q} = \frac{V^2}{2\mu K_y^2}\left(C_{m,b}\right) + \frac{I_z - I_x}{I_y} p\, r - \frac{I_E\, \varphi_E}{I_y} r + \frac{M_{other}}{I_y}$$

$$\dot{r} = \frac{V^2}{2\mu K_z^2}\left(C_n\right) + \frac{I_x - I_y}{I_z} p\, q + \frac{I_E\, \varphi_E}{I_z} q + \frac{N_{other}}{I_z}$$

Term 1. Modify Aerodynamic moments: a. with flight controls,
b. configuration changes (gear, flaps, strakes).

Term 2. Reposition the aircraft attitude on the spin axis.

Term 3. Variations in engine power.

Term 4. Spin chutes, spin rockets, etc.

Conventional means of spin recovery use flight controls to alter the aerodynamic moments (C_l, $C_{m,b}$, and C_n); configuration changes are seldom used to accomplish spin

recovery. The all-important question is "How should the flight controls be used to recover from a PSG or a spin?"

10.3.8.3 USE OF LONGITUDINAL CONTROL

The longitudinal control surface can only be effective if it can drive the angle of attack below α_s. Rarely is the elevator capable of producing this much change in pitching moment in a fully developed spin, but its use during a PSG or the incipient phase of a spin may well reduce angle of attack sufficiently. However, forward stick during a fully developed upright spin will merely cause many spin modes to progress to a higher rotation rate, which is also usually flatter. Model tests and computer studies should thoroughly investigate this control movement before it is recommended to the test pilot. Then a thorough flight test program must be conducted to confirm these predictions before such a recommendation is passed on to operational users.

10.3.8.4 USE OF RUDDER

Considering only the alteration of C_l, $C_{m,b}$, or C_n by deflection of the appropriate control surfaces, the use of rudder to change C_n has proven to be the most effective in recovering from a developed spin. Rudder deflection, if the rudder is not blanked out, produces a reduction in yaw rate which persists. The reduction in yaw rate reduces the inertia pitching couple and the angle of attack consequently decreases. Once the rotation rate has been reduced sufficiently, the longitudinal control can be used to reduce angle of attack below α_s.

Notice that the use of ailerons to produce an anti-spin rolling moment has not been discussed. Generally, in stalled flight the ailerons are not effective in producing yawing moments of any significance, though they can still be the primary anti-spin control by causing a small change in bank angle and thereby reorienting the aircraft attitude on the spin axis so that the inertial terms operate to cause recovery.

10.3.8.5 USE OF INERTIAL MOMENTS

By using ailerons to reorient the aircraft attitude on the spin axis, a component of ω can be generated on the y body axis, creating pitch rate, q. Pitch rate can then cause aircraft inertial moments to affect roll and yaw acceleration. Determining how the ailerons should be applied to reorient the aircraft attitude depends upon the relative magnitude of I_x and I_y. This can be seen from the roll and yaw acceleration equations listed below:

$$\dot{p} = \ldots + \frac{I_y - I_z}{I_x} q r \ldots \qquad 10.48$$

$$\dot{r} = \ldots + \frac{I_x - I_y}{I_z} p q \ldots \qquad 10.49$$

For instance, consider Equation 10.48 and a fuselage-loaded aircraft in a right, upright spin. $(I_y-I_z)/I_x$ is negative, while r is positive. In order to generate anti-spin roll acceleration (negative p), then q must be positive. Similarly, q must be positive to generate anti-spin yaw acceleration (Equation 10.49). For a fuselage-loaded aircraft, the pitch rate must be positive in an upright spin (right or left) to develop anti-spin yawing and rolling acceleration. Aileron applied in the direction of the spin causes the aircraft body axes to tilt so as to produce a positive component of ω along the y-axis (see Figure 10.36).

Another way to help achieve a positive pitch rate is to hold aft stick until the rotation rate begins to drop. This procedure is common in some fuselage-loaded aircraft, although it is unacceptable in others (F-104 for example). However, the most important factor is the relative sizes of I_x and I_y. Considering that $(I_x - I_y)/I_z$ is approximately six times greater for the F-104 than for the T-28, it is little wonder that aileron is a more important spin recovery control in the F-104 than is the rudder. A similar analysis of Equations 10.48 and

10.49 shows that aileron against the upright spin in a wing-loaded aircraft will produce an anti-spin yaw acceleration, but a pro-spin roll acceleration. Since wing-loaded aircraft generally spin more nose low than fuselage-loaded aircraft (with p $\cong$ r), and since they generally are recoverable with rudder and elevator, aileron-against recovery procedures are rarely recommended.

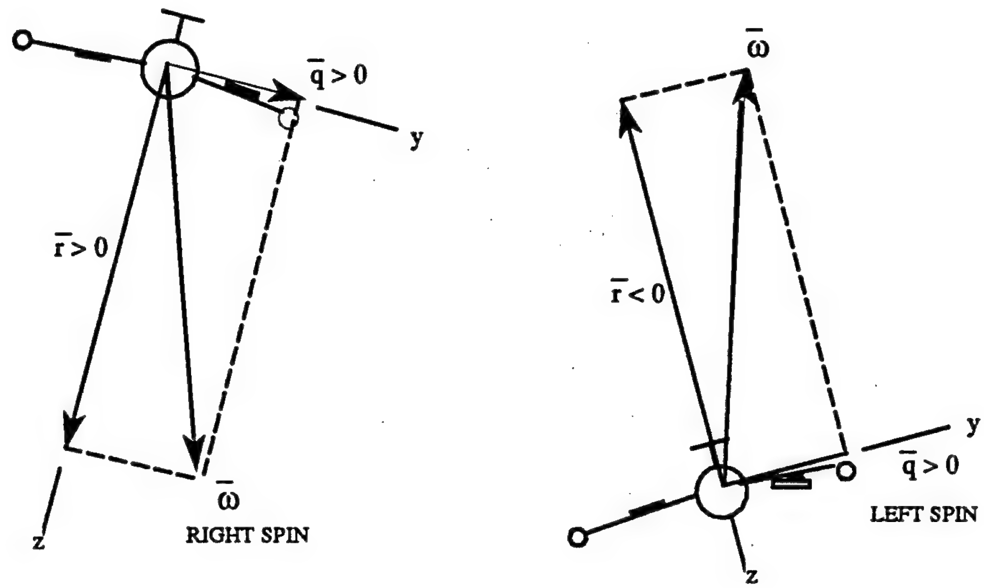


FIGURE 10.36. AILERON WITH RECOVERY PROCEDURE

10.3.8.6 OTHER RECOVERY MEANS

The other two terms which can produce anti-spin accelerations include engine gyroscopic terms and emergency recovery devices.

10.3.8.6.1 Variations in Engine Power. The gyroscopic terms are usually so small that they have little effect on recovery characteristics. Furthermore, jet engines often flame out during PSG or spin motions, particularly if the throttle is not at idle. So, although there are potential pitch and yaw accelerations available from the gyroscopic terms, NASA experience indicates that changes in engine power are generally detrimental to recovery.

10.3.8.6.2 Emergency Recovery Devices. Emergency recovery devices may take many forms - anti-spin parachutes attached to the aft fuselage, anti-spin parachutes attached to the wing tip, anti-spin rockets, strakes, etc. The design of such devices is a complex subject worthy of careful engineering in its own right. Although such design considerations are not normally the concern of test pilots, the reliability of the device, its attachments, and its jettison mechanism are of vital concern. The pilot is also likely to be concerned with tests to validate this reliability.

10.3.8.6.3 Recovery from Inverted Spins. Recovery from inverted spins is generally easier than recovery from upright spins, particularly if the rudder is in undisturbed airflow. In fact many aircraft will recover from an inverted spin as soon as the controls are neutralized. In any case rudder opposite to the turn needle may be recommended, often in conjunction with aft stick. Some fuselage-loaded T-tailed aircraft may require anti-spin aileron. An analysis of Equations 10.48 and 10.49 shows that in an inverted spin aileron against the spin is the correct anti-spin control for a fuselage-loaded aircraft.

10.3.9 SPIN THEORY REVIEW

A brief review of some basic assumptions and prerequisites is in order. For a stabilized spin to exist, there must be a balance of moments (in particular, pitching moments). Therefore the sum of the inertial pitching moment (M_i) and the aerodynamic pitching moment (M_a) must equal zero. For equilibrium, all accelerations must also be equal to zero, i.e., $\dot{p} = \dot{q} = \dot{r} = \dot{\omega} = \dot{v} = 0$. For simplicity, assume a wings level spin with no sideslip (i.e., $q = 0 = \beta$; in reality, these are mutually exclusive assumptions). Since for an erect spin M_i is always positive (nose-up), it follows from the first assumption that M_a must therefore always be negative (nose-down). For this to occur, $C_m < 0$. As a corollary to this prerequisite, for stability in the spin along the pitch axis, $C_{m\alpha}$ must also be negative, otherwise the aircraft would pitch itself up and out of the spin (as α increased, M_a and M_i would both increase). Two

other important prerequisites discussed in the spin course are the requirements to have $\alpha > \alpha_s$ and a sustained yaw rate. Summarizing these assumptions and prerequisites for a stabilized, wings level spin:

a. $M_i + M_a = 0$
b. $\dot{p} = \dot{q} = \dot{r} = \dot{\omega} = \dot{v} = 0$
c. $q = 0$
d. $\beta = 0$
e. $C_m < 0$
f. $C_{m\alpha} < 0$
g. $\alpha > \alpha_S$
h. Sustained Yaw Rate

As shown in Figure 10.37, assume an aircraft in a stabilized, erect spin to the right. Resolving the spin vector, ω, into body axis rotations gives

$$p = \omega \cos \alpha \qquad 10.50$$

$$r = \omega \sin \alpha \qquad 10.51$$

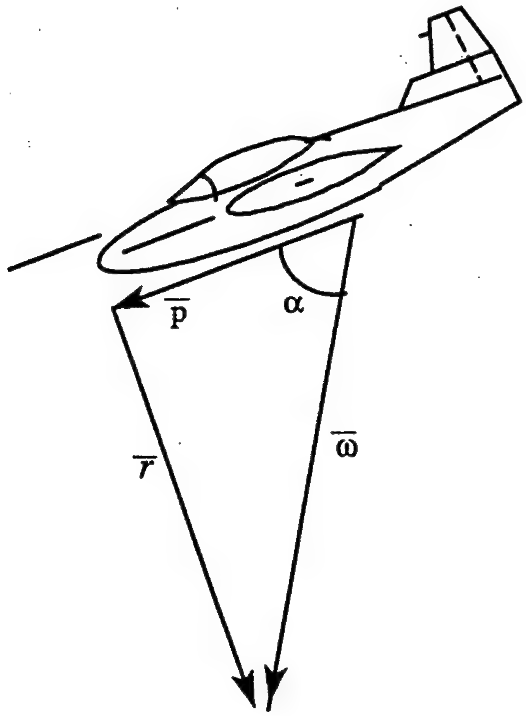


FIGURE 10.37. RESOLUTION OF SPIN VECTOR, ω

The three equations of motion used to describe an aircraft in a spin are:

$$G_x = \dot{p}I_x - qr\,(I_y - I_z) \tag{10.45}$$

$$G_y = \dot{q}I_y - rp\,(I_z - I_x) \tag{10.46}$$

$$G_z = \dot{r}I_z - pq\,(I_x - I_y) \tag{10.47}$$

where principal body axes have been assumed (i.e., $I_{xy} = I_{xz} = I_{yx} = 0$). G_x, G_y, and G_z are applied moments only and are therefore produced by aerodynamic moments L_A, M_A and N_A, respectively. Substituting and using the fact that $\dot{p} = \dot{q} = \dot{r} = 0$ for a stabilized spin yields:

$$L_a = -qr\,(I_y - I_z) \tag{10.54}$$

$$M_a = -rp\,(I_z - I_x) \tag{10.55}$$

$$N_a = -pq\,(I_x - I_y) \tag{10.56}$$

An examination of the pitch equations, $M_a = -rp(I_z - I_x)$, is in order. From the first assumption ($M_a = -M_i$) it is obvious that $M_i = rp\ (I_z - I_x)$. Substituting for r, p, the components of ω, $M_i = (\omega \sin \alpha)(\omega \cos \alpha)(I_z - I_x)$. Using the trigonometric identity, $\sin 2r = 2 \sin \Theta \cos \Theta$,

$$M_i = \frac{\omega^2}{2}(\sin 2\alpha)(I_z - I_x) = \omega^2 \left(\frac{I_z - I_x}{2}\right) \sin 2\alpha \qquad 10.57$$

Plotting $-M_i$ vs α, it is apparent that a curve, whose amplitude depends on the value $\omega^2 (I_z - I_x)/2$ is obtained (Figure 10.39). (Intuitively, the strength of the inertial pitching moment is expected to increase as ω increases.):

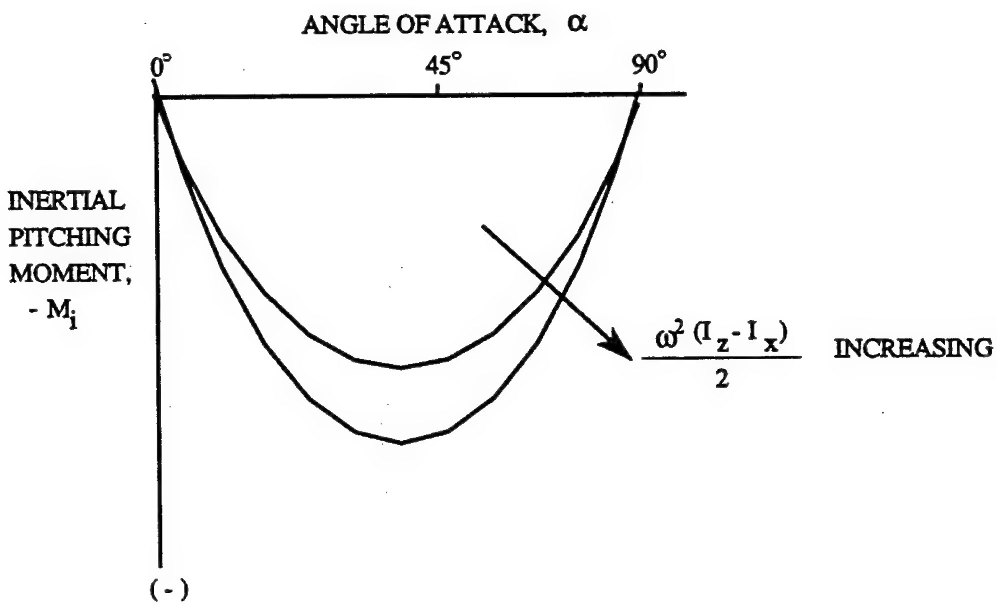


FIGURE 10.39. INERTIAL PITCHING MOMENT

For a stabilized erect spin of a "normal" aircraft, there are two prerequisites

$C_m < 0$ and $C_{m\alpha} < 0$

Examining M_a gives $M_a = C_M (1/2) \rho V^2 Sc$. Therefore, a plot of M_a vs α yields Figure 10.40.

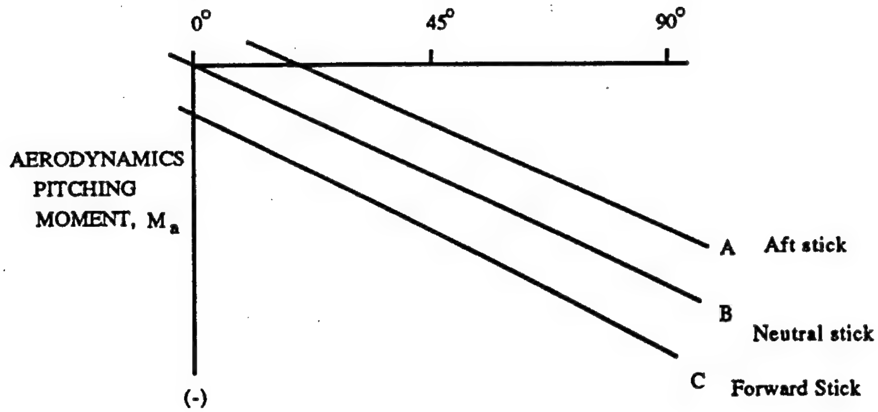


FIGURE 10.40. AERODYNAMIC PITCHING MOMENTS

Line A represents full aft stick. Line B represents neutral stick and Line C represents full forward stick. (Note: In the A-37, Curve B lies closer to Line C than to Line A.) Superimposing the M_a and $-M_i$ curves vs α, the value of α where a stabilized erect spin can occur can be located, i.e., where $M_a = - M_i$ (Figure 10.41).

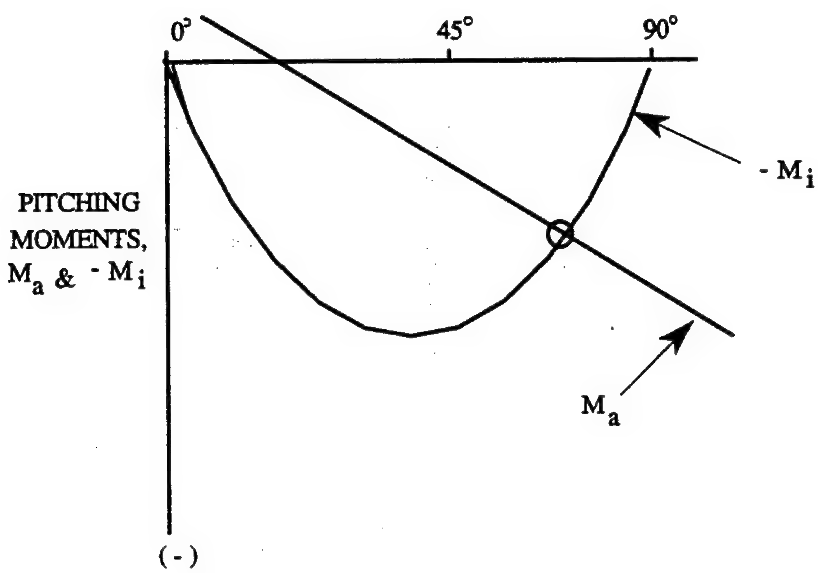


FIGURE 10.41. INERTIAL AND AERODYNAMIC PITCHING MOMENTS

At first glance, one might suspect an infinite number of values for α where a spin could occur, depending on how M_i is drawn which, in turn, depends on the choice of a value of ω, the spin rate. However, the pitch axis is not the only axis involved and consideration must be given to the yaw and roll axis stabilities to determine the value of α and ω for a stabilized spin. That analysis is conducted using rotary balance wind tunnel data and is not considered here. The purpose here is to examine the effects on the spinning aircraft once the stabilized conditions are known.

The statement, "flatter spins spin faster", bears examination. Intuitively, as the spin becomes flatter (i.e., α increases) a more nose-down moment is created due to M_a. To remain stabilized the aircraft must spin faster to generate a large nose-up moment from M_i. This can also be seen by referring to Figure 10.42.

Assuming the M_a curve doesn't change, i.e., holding the stick fixed, the spin occurs at point A for one value of ω and α. If the spin rate can be stabilized at a higher rate (i.e., if it can be propelled in some manner to a faster rate) then, without changing M_a, the aircraft must flatten out to point B. Also, if the spin became flatter for whatever reason, a new and higher spin rate would develop to compensate. Hence, the faster the spin, the flatter the spin and vice versa.

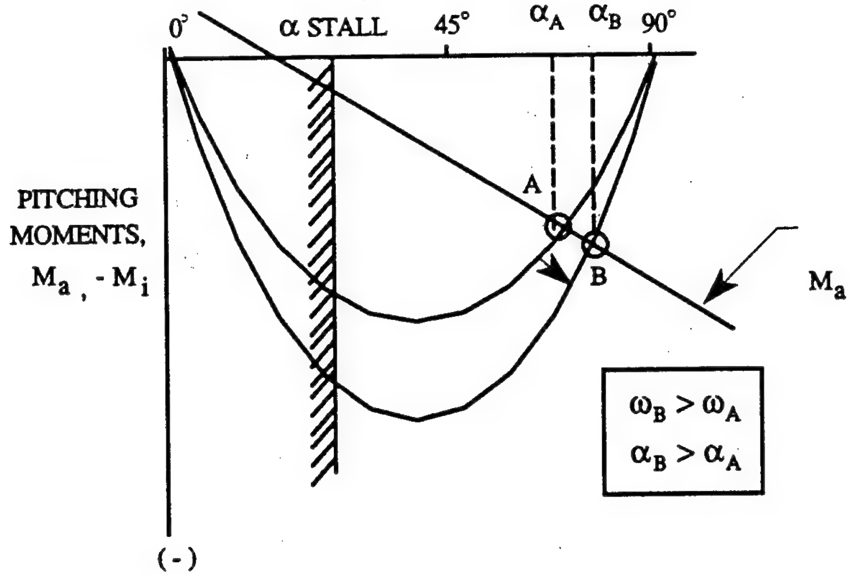


FIGURE 10.42. EFFECT OF ANGLE OF ATTACK ON SPIN RATE

Next, the effect of forward stick on the spin mode must be examined. First of all, two points must be considered: (1) the elevator remains effective in the A-37 for generating some pitching moment and (2) the aircraft seeks to maintain the equilibrium angle of attack existing before the control input. An explanation of the effects is shown in Figure 10.43.

Two curves are shown depicting M_a vs α for full aft and full forward stick. During an erect spin in the A-37, the stick is held full aft and thus the spin occurs at Point A. If the stick is moved slowly to the full forward position, the aircraft will seek a new equilibrium spin mode. The aircraft tries to maintain the initial AOA (α_A), but because the elevator is still somewhat effective, α can be reduced only a slight amount. Because more nose-down aerodynamic pitching moment has been generated by moving the stick full forward, (even though α has decreased slightly) the inertial pitching moment must also increase in magnitude in order to reach equilibrium. But M_i can only increase by allowing the spin rate, ω, to increase. The elevator is not effective enough to decrease α all the way to α_c, which is still on the original spin rate curve. Therefore, α decreases only slightly to α_B and the aircraft must spin faster to compensate, i.e., at Point B. Note that now there is a seeming contradic-

tion, i.e., the aircraft is spinning steeper and faster, however, this is due to the fact that the M_a curve along which the aircraft must operate, changed. In the previous discussion (flat and fast) the curve was held constant. Summarizing this effect, full forward stick will create the fastest spin mode at a lower α and full aft stick will create the slowest spin mode at a higher α. This explains why aft stick is important in a recovery; it slows the spin down (along with opposite rudder) thereby reducing M_i to a point where the elevator has enough authority to overcome it (i.e., $|M_a| > |M_i|$) and pitch the aircraft down out of the spin.

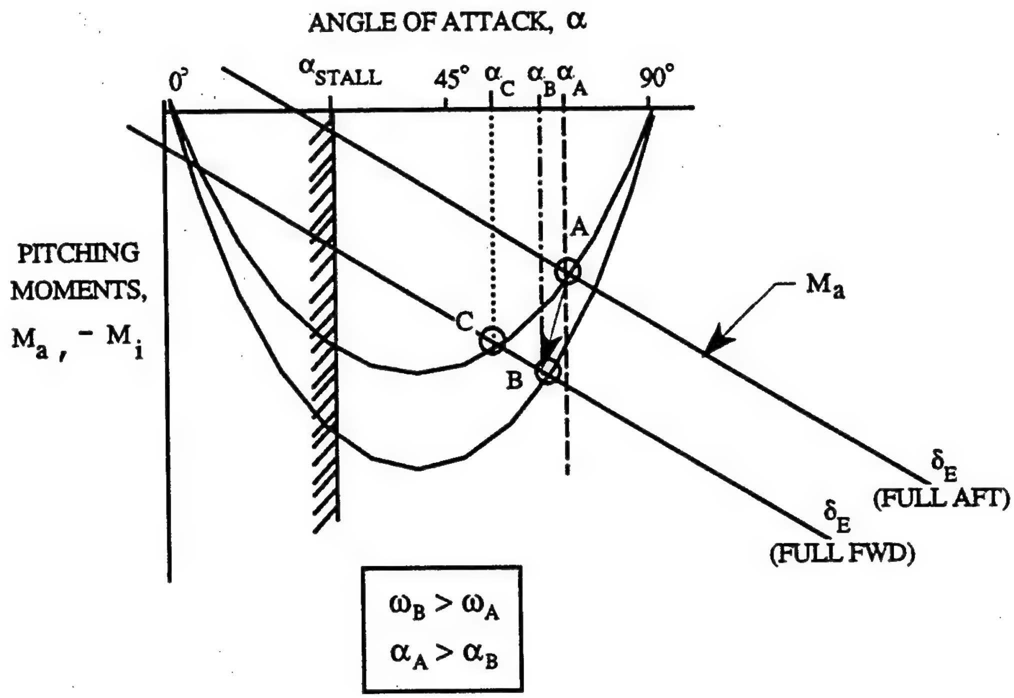


FIGURE 10.43. EFFECT OF STICK POSITION ON SPIN RATE

Finally, the use of ailerons in a spin must be examined. To do this it must be understood that use of the ailerons uses roll to reorient the aircraft along the spin axis slightly, thereby creating a pitch rate, q. This pitch rate then couples in the equations of motion to create a yaw acceleration or deceleration, depending on the aircraft's inertial loading. Ailerons with the turn needle always create a positive pitch rate, regardless if the

aircraft is erect or inverted. The equation of motion of importance is the yaw acceleration equation

$$\dot{r} = \frac{N_a}{I_z} + pq\left(\frac{I_x - I_y}{I_z}\right)$$

For a wing-loaded aircraft like the A-37, $I_x > I_y$. Considering an erect spin to the right, r and p are positive and aileron with the turn needle (i.e., right aileron) causes q to be positive. Pro-spin rudder is held until recovery, thus N_a is positive. Therefore, r is positive and is accelerating since $\dot{r}$ is also positive. From the previous discussion (faster/flatter), since putting in aileron increases ω but has no appreciable effect on the M_a curve, α must therefore increase as aileron is applied in the direction of the spin. Typically in the A-37, aileron with the spin will increase the yaw rate 10-15% and α will increase approximately 10°.

Admittedly, this is a simplified analysis. There are many more complex interrelationships occurring in the other two axes of motion in much the same way as in the pitch axis. However, this simplified view is still sound and should promote understanding of what occurs in a stabilized spin.

10.4 HIGH ANGLE OF ATTACK FLIGHT TESTS

10.4.1 STALL FLIGHT TESTS

Stalls, a familiar maneuver mastered by every pilot when first learning to fly, must not be taken for granted in a test program. There is a rather large collection of examples from flight test history to document the need for caution. Designs that combine an inherent

pitchup tendency with measurable spin characteristics have contributed much to these examples. Stalls are usually first demonstrated by a contractor pilot, but it is possible for a military test pilot to find himself doing the first stalls in a particular configuration, especially on test bed research programs where frequent modifications and changes are made after the vehicle has been delivered by the contractor.

The cautious approach starts with good preplanning. Discussion with the appropriate engineering talent of the predicted stall characteristics, and development of the most promising recovery technique for each stage of the stall, including possible post-stall gyrations is a necessity. In marginal cases, a suggestion for further wind tunnel testing or other alternative investigations might be warranted. The most favorable loading and configuration to be used in the initial stages must be determined. Stall and spin practice in trainer aircraft will enhance pilot performance during any out-of-control situations that might develop.

If pitchup or other control problems seem remotely possible, the first runs should terminate early in the approach to the stall and the data carefully examined (on the ground) for trends such as lightening or reversal of control, excessive attitudes, or sink rates. Advancing this data systematically on subsequent flights and avoiding the mistake of suddenly deciding in flight, because things are going well, to take a bigger step than planned is a necessity. Stall characteristics must be evaluated in relation to their influence on mission accomplishment. Thus, both normal and accelerated stalls must be performed under entry conditions which could result from various mission tasks. However, prior to evaluating stalls entered from these conditions, a more controlled testing approach should be employed. This approach allows lower deceleration rates into the stall and lower pitch attitudes at the stall, thereby reducing chances for "deep-stall" penetration without adequate buildup. After the controlled stall investigation, if stall characteristics permit, simulated inadvertent stalls should be investigated under conditions representative of operational procedures.

10.4.2 THE CONTROLLED STALL TEST TECHNIQUE

The easiest and safest approach to controlled stall testing is to divide the investigation into three distinct parts:

1. Approach to the stall
2. Fully developed stall
3. Stall recovery

10.4.2.1 APPROACH TO STALL

During this phase of the investigation, adequacy of stall warning and retention of reasonable airplane controllability are the primary items of interest. Assessment of stall warning requires subjective judgement by the pilot. Only the pilot can decide when he has been adequately warned. Warning must occur sufficiently in advance of the stall to allow prevention of the stall by normal control applications after a reasonable pilot reaction time. However, stall warning should not occur too far in advance of the stall. For example, it is essential that stall warning for approach configuration occur below normal approach speed. Reference 10.3 specifies definite upper and lower airspeed limits within which warning should occur. Stall warning which occurs too early is not only annoying to the pilot but is meaningless as an indication of proximity to the stall.

The type of stall warning is very important. Primary stall warning is generally in the form of airframe buffet, control shaking, or small amplitude airplane oscillations in roll, yaw, or pitch. Other secondary cues to the approach of the stall may be high pitch attitude, large longitudinal control pull forces (of course, this cue can be destroyed by "trimming into the stall"), large control deflections or sluggish control response. In any case, stall warning, whether natural or artificial, should be unmistakable, even under conditions of high pilot workload and stress and under conditions of atmospheric turbulence. If an artificial stall warning device is installed, approach to the stall should be evaluated with the device operative and inoperative to determine if the device is really required for normal operations.

During this phase of the evaluation, the test pilot must evaluate stall warning with the intended use and operational environment in mind. He must remember that he is specifically looking for the stall warning under controlled conditions. The operational pilot probably will

not be. Will the operational pilot, preoccupied by other tasks and not concentrating on stalls, recognize the approach of a stall and be able to prevent it?

The general flying qualities of the airplane should be investigated during the approach to the stall as well as stall warning characteristics. Longitudinal, lateral, and directional control effectiveness for maintaining a desired attitude may deteriorate significantly during the approach to the stall. Loss of control about any axis such as uncontrollable pitch-up or pitch-down, "wing drop," or directional "slicing" may define the actual stall. During the approach to the stall, the test pilot should be particularly aware of the amount of longitudinal nose-down control available because of the obvious influence of this characteristic on the ability to "break" the stalled condition and make a successful recovery.

The approach to Stall Phase of testing usually begins with onset of stall warning and ends at the stall, therefore the test pilot will certainly be concerned with the manner in which the airplane stalls and the ease of recovery. However, primary emphasis is placed on obtaining an accurate assessment of stall warning and general flying qualities during the approach to the stall. During initial investigations, it may be prudent to terminate the approach short of the actual stall, penetrating deeper and deeper with each succeeding approach until limiting conditions or the actual stall are reached. In addition, the rate of approach should be low initially, approximately one knot per second for normal stalls and two knots per second for accelerated stalls. As experience is gained, deeper penetrations at faster deceleration rates must be performed unless safety considerations dictate otherwise.

The test pilot should record at least the following data during the approach to the stall:

a. Airspeed, angle of attack, and altitude at stall warning

b. Type and adequacy of stall warning

c. Longitudinal control force at stall warning (either measured or estimated)

d. Qualitative comments regarding controllability and control effectiveness

e. Aircraft weight

10.4.2.2 FULLY DEVELOPED STALL

Stall has been defined as the minimum steady speed attainable, or usable, in flight. This minimum may be set by a variety of factors, for example:

a. Reaching C_{Lmax} - the conventional stall

b. Insufficient longitudinal control to further decrease speed, or increase α - lack of elevator power

c. Onset of control problems (uncommanded motion about any axis)
   (1) Pitchup
   (2) Insufficient lateral-directional control to maintain attitude
   (3) Poor dynamic characteristics

d. Back-side problems
   (1) High sink rate
   (2) Insufficient wave-off capability
   (3) Excessive pitch attitude

During the fully-developed Stall Phase of testing, the primary objective is to accurately define the stall and the associated airplane behavior. The stall should be well-marked by some characteristic, such as pitch-up or pitch-down or lateral or directional divergence. In general, any pitch-up or directional divergence at the stall is undesirable because pitch-up may precipitate a deep stall penetration and directional divergence may lead to a spin. Pitch-down at the stall and lateral divergence may be acceptable. However, severe rolling, pitching, or yawing or any combination of the three are obviously poor characteristics.

Control effectiveness as evidenced by the pilot's ability to control or induce roll, pitch, or yaw should be evaluated in the stall, if airplane behavior permits this to be done safely. Obviously, control effectiveness should be evaluated with a suitable build-up program. Initially, control inputs only large enough to effect an immediate coordinated recovery should be used. As experience is gained, the airplane should be maintained in the stalled condition for longer and longer periods of time, and the effectiveness of all controls evaluated with larger and larger control deflections.

Actual flight test techniques to be used during stall testing must be agreed upon by the contractor, the System Program Office (SPO) and the flight test center performing the tests.

Two flight test methods for defining the stall and the associated airplane behavior are presented below.

10.4.2.2.1 Level Flightpath Method. This method, involving a level flightpath, is an older method that is valid only for unaccelerated stalls. It has several disadvantages that limit its application, but in certain cases such as VSTOL testing or initial envelope expansion it might prove useful. It has been largely replaced by the second method that involves a curved flightpath and is valid for both accelerated and unaccelerated stalls (Figure 10.44).

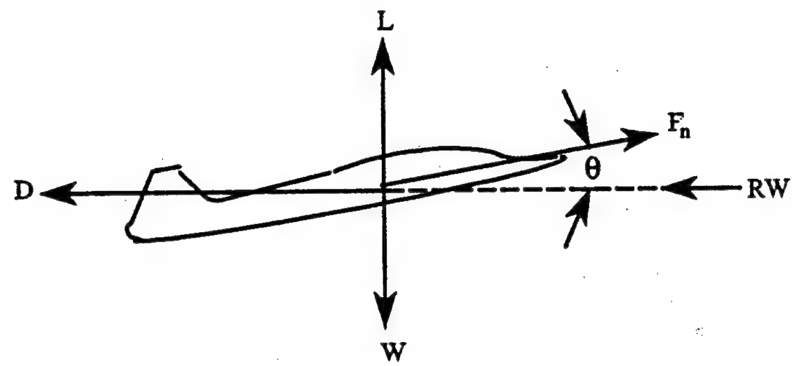


FIGURE 10.44. LEVEL FLIGHT PATH METHOD

$L + F_n \sin \Theta = W$ and the flightpath is straight. In order to slow the aircraft to stall speed, however, an acceleration (a_D) in the drag direction must be obtained by adjustment of thrust or drag such that D is greater than $F_n \cos \Theta$. This represents a disadvantage of the method, since a particular trim power or drag configuration cannot be maintained to the stall.

10.4.2.2.2 Curved Flightpath Method. MIL-STD 1797, Paragraph 3.4.2, requires that stall speed (V_s) be defined at 1-g normal to the flightpath, and that the aircraft be initially trimmed at approximately 1.2 V_s, after which trim and throttle settings remain constant. To achieve these requirements, once the trimmed conditions have been set the aircraft pitch attitude is increased to achieve a slight climb (less than 500 FPM), so as to initiate a bleed rate of one or two knots per second. Experience has shown that undesirable dynamic effects are encountered if bleed rates much in excess of one or two knots per second are used. Based on experience, arbitrary maximum bleed rates of one knot per second for unaccelerated stalls, and two knots per second for accelerated stalls have been set. Once the initial climb has been established, pitch attitude is controlled so as to maintain or increase the rate of climb (1500

FPM maximum). This technique conservatively assures a stall speed at 1-g normal to the flightpath. Stall speeds occurring at load factors other than 1-g normal to the flightpath shall be corrected as described in MIL-STD 1797, Paragraph 3.4.2. A tolerance of $\pm$ 500 feet from desired stall altitude is allowed.

The test pilot should strive to record at least the following data regarding the stall:

1. Airspeed, angle of attack, and altitude at stall
2. Load Factor
3. Characteristic which defines the stall
4. Longitudinal control force at the stall (either measured or estimated). The ratio of longitudinal control forces at stall and stall warning is a rough indication of longitudinal stability in the high angle of attack region and an indication of the ease of inadvertent stalling
5. Qualitative descriptive comments
6. Aircraft weight

10.4.2.3 STALL RECOVERY

During this phase of the investigation, primary items of interest are the ease of recovery (the pilot's task), general flying qualities during the recovery, altitude required for recovery and the determination of an optimum recovery technique.

The recovery is started when the stall or minimum steady speed has been attained. For a conventional stall this is indicated by the inability to maintain the desired load factor -- usually a sudden break is apparent on the cockpit accelerometer.

The goal of the recovery must be specified. For example, the goal of recovery for configurations commensurate with combat maneuvering may be to regain sufficient control effectiveness about all three axes to perform offensive or defensive maneuvering tasks; the attainment of level flight may not be critical in these configurations. The goal of recovery for takeoff and approach configurations should be attainment of level flight with a minimum

loss of altitude and the regaining of sufficient control effectiveness to safely maintain stall-free conditions. In each case, the test pilot must clearly define "stall recovery."

In a test program, all promising recovery procedures consistent with the objectives should be tried. It is important to have the recovery specified in detail before each drill -- not to wait until the stall breaks to decide what procedure is to be used. There are no iron-clad rules for recovery -- a "standard procedure" such as full military power could be disastrous in certain vehicles. Keep the instrumentation running throughout the recovery until the goal has been attained. In the case of minimum altitude loss, this would be when rate of descent is zero and the aircraft is under control (the altimeter is the first indication of ROC = 0).

During initial investigation, the stall recovery procedures specified in pertinent publications should be utilized and the ease of effecting recovery evaluated. If no procedure has been developed, initial recovery must be accomplished with a "preliminary" technique formulated from all available technical information. As experience is gained, various modifications to the recovery procedure should be made until an optimum procedure is determined. In arriving at an optimum procedure for use by the operational pilot, the test pilot must not only consider the effectiveness of the technique (in terms of altitude lost or maneuverability regained), but must also consider the simplicity of the technique.

The test pilot should record at least the following data regarding stall recovery:

1. Qualitative comments on ease of recovery
2. Optimum recovery technique
3. Altitude lost in recovery
4. Qualitative comments on control effectiveness

10.4.3 SPIN FLIGHT TESTS

10.4.3.1 SPIN PROJECT PILOTS BACKGROUND REQUIREMENTS

Under current stall/post-stall/spin demonstration specification (MIL-F-83691B, Paragraph 3.3), military pilots will participate on high angle of attack investigations concur-

rently with the contractor's pilots. Therefore, it is imperative that the military test pilots assigned to a high angle of attack investigation be thoroughly familiar with all available background information concerning the investigation. This paragraph summarizes the preparation required for post-stall/spin investigations. The discussion is purposely general in nature; it will not specifically address the tests flown in the curriculum at the USAF Test Pilot School. These flights are described in detail in the current Flying Qualities Phase Planning Guide.

The methods available to the modern test pilot for pre-flight spin test research are:

1. Conventional wind tunnel

2. Dynamic models

   a) Wind tunnel free flight model
   b) Radio controlled model

3. Vertical spin tunnel

4. Rotary balance

5. Simulator

10.4.3.1.1 Conventional Wind Tunnel. Literature research should begin with the best and most current wind tunnel data available. Take careful note of any configuration or mass changes which were made since the available wind tunnel data were obtained. Use a critical eye when looking at the angle of attack and angle of sideslip ranges tested in the tunnel. Go over this data very carefully with the flight test engineers and try to ascertain the probable spin modes and optimum recovery techniques for each of them, as well as the optimum recovery procedure for post-stall gyrations, if one is known. Start looking, even at this stage, for the simplest recovery technique possible. If possible, obtain analytical data to confirm or deny the possibility of using a common recovery procedure for both post-stall gyrations and spins (MIL-F-83691B, Paragraph 3.4.3). Spin test reports of similar aircraft should be reviewed thoroughly, but care must be exercised in extrapolating results. The spin characteristics of aircraft which are quite similar in appearance can vary drastically. Attempt to predict the

effect that various loadings and configurations will have on post-stall/spin characteristics so that initial tests can be planned conservatively. As examples, the A-1 has loadings from which recovery is not acceptable (10.13:6) and highly asymmetric loadings in the A-7D may prolong recovery to an unacceptable degree. Flight tests of the A-7D were not performed with loadings of greater than 13,000 foot-pounds of asymmetry (10.14:11).

10.4.3.1.2 Dynamic Model Techniques (10.15:13-2). As a result of the complexity of the stall/spin problem, and the lack of proven alternate predictive methods, the most reliable source of information on stall/spin characteristics prior to actual flight tests of the particular airplane has been tests of dynamically scaled airplane models. A properly scaled dynamic model may be thought of as a simulator with the proper values of the various aerodynamic and inertial parameters.

Several unique dynamic model test techniques for stall/spin studies have been developed including: (1) the wind-tunnel free-flight technique, (2) the outdoor radio-controlled model technique, and (3) the spin-tunnel test technique.

10.4.3.1.2.1 Model Scaling Considerations (10.15:13-2). Dynamic models must be scaled in each of the fundamental units of mass, length, and time in order to provide test results that are directly applicable to the corresponding full-scale airplane at a given altitude and loading condition. As a result of scaling, the motions of the model are geometrically similar to those of the full-scale airplane and motion parameters can also be scaled.

Some limitations of the dynamic model test techniques are apparent. For example, the model is tested at a value of Reynolds number considerably less than those of the full-scale airplane at comparable flight conditions. Although the linear velocities of the model are smaller than full-scale values, the angular velocities are greater than full-scale values.

The discrepancy in Reynolds number between model and full-scale airplane can be an important factor which requires special consideration for stall/spin tests. During spin-tunnel tests, large Reynolds number effects may be present which cause the model to exhibit markedly different characteristics than those associated with correct values of Reynolds number.

The fact that the angular velocities of the model are much faster than those of the airplane poses special problems with regard to controllability of the model for certain

techniques. Because the human pilot has a certain minimum response time, it has been found that a single human pilot cannot satisfactorily control and evaluate dynamic flight models.

The stall and spin of an airplane involve complicated balances between the aerodynamic and inertial forces and moments acting on the vehicle. In order to conduct meaningful tests with dynamic models, it is important that these parameters be properly scaled. Simply scaling dimensional characteristics without regard to other parameters will produce erroneous and completely misleading results.

As a result of the shortcomings of theoretical methods, the most reliable source of information on stall/spin characteristics prior to full-scale flight tests has been tests of dynamically scaled models.

10.4.3.1.2.2 The Wind-Tunnel Free-Flight Technique (10.15:13-2). The wind-tunnel free-flight technique is used specifically to provide information on flight characteristics for angles of attack up to and including the stall. The test setup for this model test technique is illustrated by the sketch shown in Figure 10.45. Two pilots are used during the free-flight tests. One pilot controls the longitudinal motions of the model. The second pilot controls the lateral-directional motions of the model. The model is powered by compressed air, and the level of thrust is controlled by a power operator. The human pilots do not sense accelerations as the pilot of an airplane does, and must therefore, fly with sight cues as the primary source of information.

The cable attached to the model serves two purposes. The first purpose is to supply the model with compressed air, electric power for control actuators, and control signals. The second purpose of the cable is concerned with safety. A portion of the cable is a steel cable that passes through a pulley above the test section. This part of the flight cable is used to catch the model when a test is terminated or when an uncontrollable motion occurs. The entire flight cable is kept slack during the flight tests by a safety-cable operator who accomplishes this job with a high-speed pneumatic winch.

The model incorporates limited instrumentation for measurements of motion and control deflections.

The wind-tunnel free-flight technique can produce valuable information during studies of flight motions at high angles of attack and at the stall. Various phases of a typical

investigation would include: (1) flights at several angles of attack up to and including the stall to evaluate dynamic stability characteristics, (2) an evaluation of pilot lateral control techniques at high angles of attack, and (3) an evaluation of the effects of stability augmentation systems. The wind-tunnel free-flight technique has several inherent advantages: (1) because the tests are conducted indoors, the test schedule is not subject to weather conditions; (2) the tests are conducted under controlled conditions and a large number of tests can be accomplished in a relatively short period of time; (3) airframe modifications are quickly evaluated and (4) models used in the technique are relatively large (11:1-scale for most fighter configurations) and can, therefore, be used in force tests to obtain static and dynamic aerodynamic characteristics for analysis of the model motions and as inputs for other forms of analysis, such as piloted simulators.

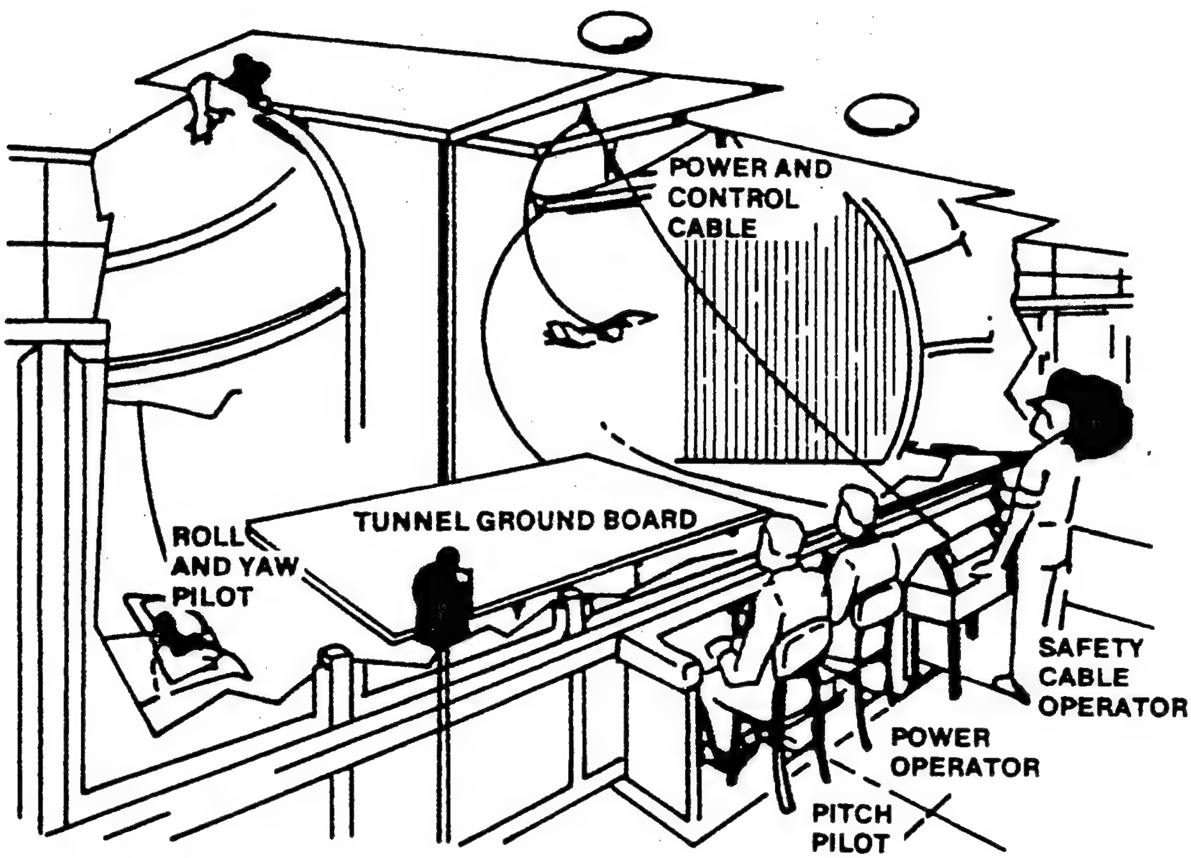


FIGURE 10.45. TEST SETUP FOR WIND-TUNNEL FREE-FLIGHT TESTS

10.4.3.1.2.3 The Outdoor Radio-Controlled Model Technique. (10.15:13-3, 4). A significant void of information exists between the results produced by the wind-tunnel free-flight test technique for angles of attack up to and including the stall, and the results produced by the spin-tunnel test technique, which defines developed spin and spin-recovery characteristics. The outdoor radio-controlled model technique has, therefore, been designed to supply information on the post-stall and spin-entry motions of airplanes. The radio-controlled model technique consists of launching an unpowered, dynamically scaled, radio-controlled model into gliding flight from a helicopter, controlling the flight of the model from the ground, and recovering the model with a parachute. A photograph showing a typical model mounted on the launching rig of a helicopter is shown in Figure 10.46.

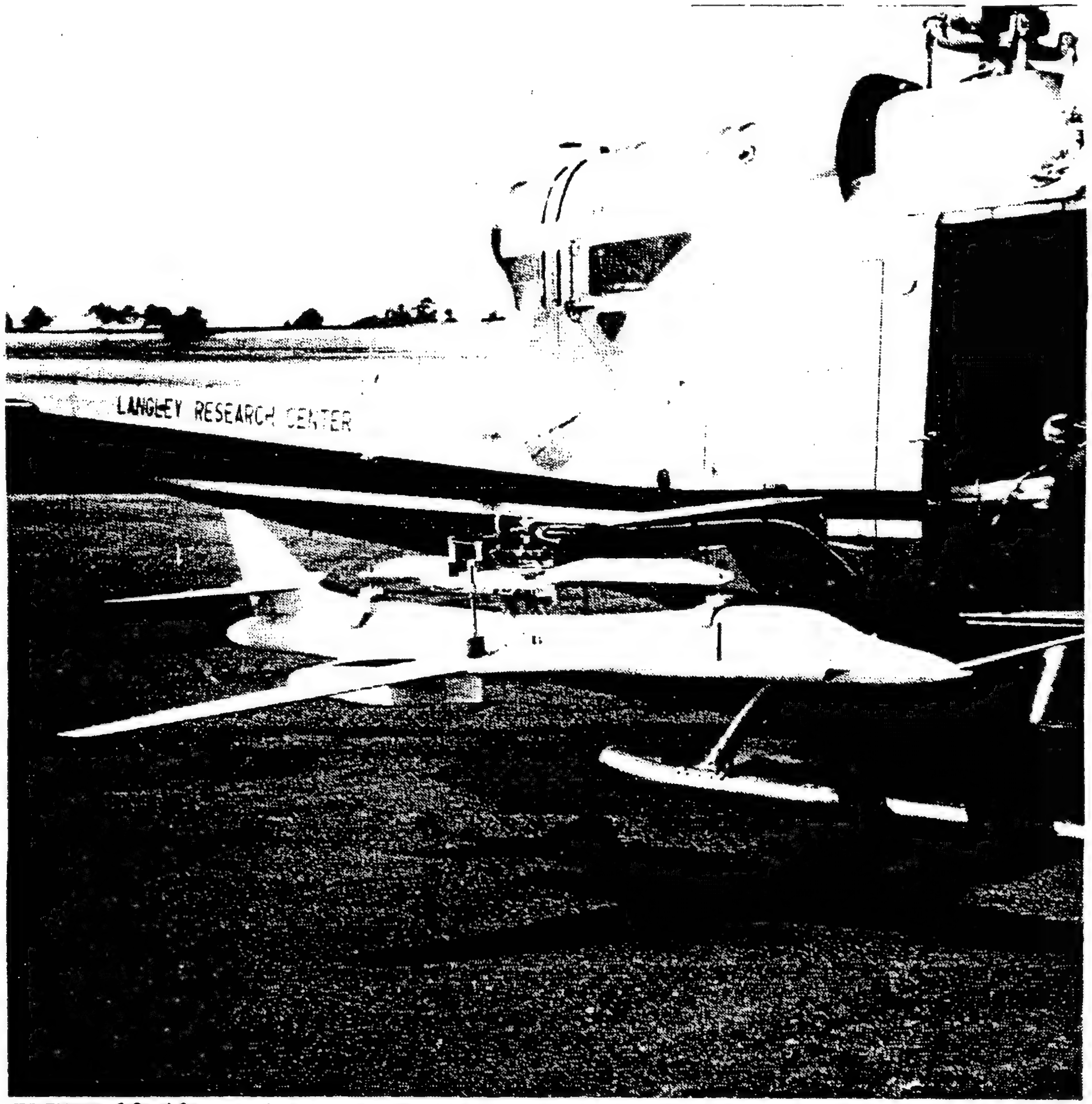


FIGURE 10.46. B-1 RADIO CONTROLLED DROP MODEL MOUNTED ON A HELICOPTER

The models used in these tests are made relatively strong to withstand high landing impact loads of 100 to 150 g's. They are constructed primarily of fiberglass plastic, with the fuselage case being thick hollow shells and the wings and tails having solid balsa cores with fiberglass sheet coverings. Radio receivers and electric actuators are installed to provide individual operation of all control surfaces and a recovery parachute. Proportional-type control systems are used in this technique.

The models are trimmed for approximately zero lift, and launched from the helicopter at an airspeed of about 40 knots and an altitude of about 5000 ft. The models are allowed to dive vertically for about five seconds, after which the horizontal tails are moved to stall the model. After the stall, various control manipulations may be used; for example, lateral-directional controls may be moved in a direction to encourage any divergence to develop into a spin. When the model has descended to an altitude of about 500 ft., a recovery parachute is deployed to effect a safe landing.

The outdoor radio-controlled model technique provides information which cannot be obtained from the other test techniques. The indoor free-flight tests, for example, will identify the existence of a directional divergence at the stall, but the test is terminated before the model enters the incipient spin. In addition, only 1-g stalls are conducted. The radio-controlled technique can be used to evaluate the effect of control inputs during the incipient spin, and accelerated stalls can be investigated. At the other end of the stall/spin spectrum, spin-tunnel tests may indicate the existence of a flat or nonrecoverable spin mode, but it may be difficult for the airplane to attain this spin mode from conventional flight - the difference being that models in the spin tunnel are launched at about 90° angle of attack with a forced spin rotation. The radio-controlled test technique determines the spin susceptibility of a given airplane by using spin entry techniques similar to that of the full-scale airplane.

The radio-controlled technique determines (1) the spin susceptibility of a configuration, (2) control techniques that tend to produce developed spins, and (3) the effectiveness of various control techniques for recovery from out-of-control conditions.

There are several limitations of the radio-controlled technique that should be kept in mind. The first is that the tests are conducted out-of-doors. The test schedule is, therefore, subject to weather conditions, and excessive winds and rain can severely curtail a program.

This technique is relatively expensive. Expensive flight instrumentation is required to record. the motions of the model; and the large size of these models requires the use of powerful and reliable electronic equipment. Costs are compounded by the fact that the model and its electronic equipment frequently suffer costly damage on landing impact. Because of this higher cost and the slow rate at which radio-controlled drop model tests can be accomplished, this technique is used only in special cases where the simpler and cheaper wind-tunnel and spin-tunnel techniques will not give adequate information; for example, when it is necessary to know whether an airplane can be flown into a particular dangerous spin mode, or when one wants to investigate recovery during the incipient spin. Conversely, most of the exploratory work, such as developing "fixes" for a departure at the stall or investigating a variety of spin-recovery techniques, is done in the wind tunnels.

10.4.3.1.3 The Spin-Tunnel Test Technique (10.15:13-4,5). The best known test technique used today to study the spin and spin-recovery characteristics of an airplane is the spin-tunnel test technique. A cross-sectional view of the NASA Langley Vertical Spin Tunnel is shown in Figure 10.47.

In this tunnel, air is drawn upward by a fan located above the test section. Models are hand-launched at about 90° angle of attack, with pre-rotation, into the vertically rising airstream. The model then seeks its own developed spin mode or modes. For recovery, the tunnel operator deflects the aerodynamic controls on the model to predetermined positions by remote control.

In a spin-tunnel investigation, the program consists of (1) determination of the various spin modes and spin-recovery characteristics, (2) study of the effect of center-of-gravity position and mass distribution, (3) determination of the effect of external stores, and (4) determination of the size and type parachute required for emergency spin recovery.

In a typical spin-tunnel test program, tests are made at the normal operating loading condition for the airplane. The spin and spin-recovery characteristics are determined for all combinations of rudder, elevator, and aileron positions for both right and left spins. In effect, a matrix of both the spin and spin-recovery characteristics are obtained for all control

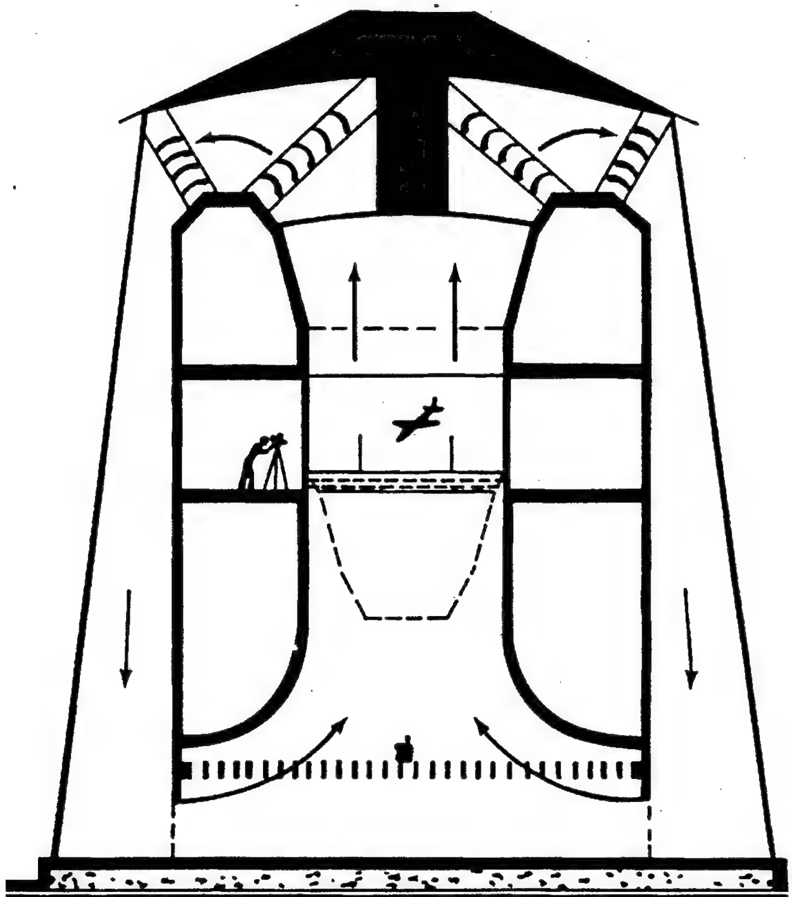


FIGURE 10.47. CROSS-SECTIONAL VIEW OF NASA LANGLEY VERTICAL SPIN TUNNEL (10.15:13-9)

settings for the normal loading operating condition. Using these data as a baseline, selected spin conditions are treated again with incremental changes to the center of gravity and/or mass conditions. Then, based on the effects of these incremental changes, an analysis is made to determine the spin and spin-recovery characteristics that the corresponding airplane is expected to have. Also, the effectiveness of various control positions and deflections are analyzed to determine which control techniques are most effective for recovery. After the spin-recovery characteristics for the normal loading conditions have been determined, additional tests are made to determine the effects of other loading conditions, store configurations (including asymmetric stores), and other items of interest such as speed brakes and leading and trailing-edge flaps.

The parachute size required for emergency spin recovery is determined for the most critical spin conditions observed in the spin-tunnel tests, and is checked at other conditions throughout the test program. If the parachute size is found to be too small for other conditions, the size is adjusted so that the parachute finally recommended for use on the spin demonstration airplane will be sufficient to handle the most critical spins possible on the airplane for any loading.

As a result of the combination of the relatively small scale of the model and the low tunnel speeds, spin-tunnel tests are run at a value of Reynolds number which is much lower than that for the full-scale airplane. Experience has shown that the differences in Reynolds number can have significant effects on spin characteristics displayed by models and the interpretation of these results. In particular, past results have indicated that very significant effects can be produced by air flowing across the forward fuselage at angles of attack approaching 90°. These effects are influenced by the cross-sectional shape of the fuselage forebody and may be extremely sensitive to Reynolds number variations. Particular attention is, therefore, required for documentation of this phenomenon prior to spin-tunnel tests. This evaluation has been conducted in the past with the aid of static force tests over a wide Reynolds number range.

10.4.3.1.4 Rotary-Balance Tests (10.15:15-5,6). The rotary-balance test technique has produced significant information regarding the complex aerodynamic characteristics of airplane configurations during spinning motions. Six-component measurements are made of the aerodynamic forces and moments acting on the wind-tunnel model during continuous 360° spinning motions at a constant angle of attack. Past studies identified some of the major factors which influence spin characteristics such as the autorotative tendencies of unswept wings and certain fuselage cross-sectional shapes. The characteristics of the basic configuration, the effects of individual and combined control deflections, the effects of tail surfaces and nose strakes, and the effects of spin radius and sideslip are determined. Test results identify configuration features which can have large effects on the aerodynamic spin characteristics of modern aircraft. Some pro-spin flow mechanisms have been identified.

Test results have also indicated that the aerodynamic moments (particularly yawing and pitching moments) exhibited by current military configurations vary nonlinearly with spin rate. Nonlinear moments have a large effect on calculated spin motions, and agreement is obtained with dynamic model tests for smooth, steady spins when such data are used as inputs for the calculations. On the other hand, conventional calculation techniques using conventional linearized static and dynamic stability derivatives often produce completely erroneous results.

The results of rotary-balance tests conducted for several current fighter configurations indicate that the aerodynamic characteristics of these vehicles during spins are extremely complex phenomena which tend to be Reynolds number dependent and which vary nonlinearly with spin rate. Computer studies of spinning motions have indicated that data obtained from rotary-balance tests will be required for the development of valid theoretical spin prediction techniques.

10.4.3.1.5 Simulator Studies (10.13: 15-6,7). The model test techniques previously discussed have several critical shortcomings. For example, the inputs of the human pilot have been minimized or entirely eliminated. In addition, the use of unpowered models and space constraints within the wind tunnels do not permit an evaluation of the spin susceptibility of airplanes during typical air combat maneuvers. Finally, the effects of sophisticated automatic control systems are not usually evaluated because of space limitations within the models. In order to provide this pertinent information, a piloted simulation test technique has been developed as a logical follow-on to the model tests.

Simulator application to the stall/spin area is dependent on the development of a valid mathematical model of the airplane under consideration. In view of the present lack of understanding of aerodynamic phenomena at spin attitudes, the simulation studies are currently limited to angles of attack near the stall, and fully developed spins are not simulated. Rather, the studies are directed toward an evaluation of the spin susceptibility or stall/departure characteristics of the airplane during typical air combat maneuvers and the effects of automatic control systems on these characteristics.

Simulation studies have indicated that it is an extremely valuable tool for stall/spin research. Correlation of results with those obtained from full-scale flight tests for several current fighters has indicated agreement, particularly with regard to the overall spin resistance of the configurations.

Extension of piloted simulation techniques to high angles of attack provides valuable insight as to the spin susceptibility of fighter configurations during representative air combat maneuvers. In addition, use of simulators is an effective method for the development and evaluation of automatic spin prevention concepts.

10.4.3.2 PILOT PROFICIENCY

It is imperative that the test pilot engaged in a post-stall/spin test program have recent experience in stalls and in spinning aircraft as similar as possible to the test aircraft. Obviously, such aircraft should be those cleared for intentional spins. Coupled departures in a mildly spinning aircraft may be helpful in simulating the post-stall gyrations of an aircraft not cleared for intentional departures. Lack of spin practice for as little as three months will reduce the powers of observation of even the most skilled test pilot. Therefore, he should practice until he is at ease in the post-stall/spin environment immediately prior to commencing the data program. Centrifuge rides, with simulated instrumentation procedures and required data observations, can also be useful.

10.4.3.3 CHASE PILOT/AIRCRAFT REQUIREMENTS

A highly qualified chase pilot in an aircraft compatible with the test aircraft increases the safety factor and adds another observer. The chase pilot should participate fully in the preparation phase. In fact it is preferable that more than one pilot be assigned to a given project. Not only does such an arrangement permit more than one qualitative opinion, but by alternating between post-stall/spin and chase assignments, each pilot gets at least two viewpoints. He can evaluate the post-stall/spin characteristics both as an in-the-cockpit observer and from the somewhat more detached chase position. Of course, from a flying safety viewpoint the benefits of a competent chase pilot should be in an airplane with performance compatible with that of the test aircraft. His responsibilities include: staying

close enough to observe and photograph departures, post-stall gyrations, and any spins; staying out of the way of an uncontrollable test aircraft; and being immediately in position to check any unusual circumstances such as lost panels, malfunctioning drag/spin chutes, or control surface positions. And, of course, if necessary, he can call out canopy jettison/ejection altitudes. All these responsibilities point up the importance of a well-prepared, observant chase pilot in a similar aircraft.

10.4.4 DATA REQUIREMENTS

The following two paragraphs are intended to provide only general guidance. The test plan for the specific project must be consulted for more detailed and specific requirements.

10.4.4.1 DATA TO BE COLLECTED

The flight test engineer will be primarily concerned with the required quantitative data. Rates of pitch, roll, and yaw, angular accelerations about each axis, control surface positions, angle of attack, indicated airspeed, and altitude are but a few of the typical time histories plotted meticulously by engineers. The pilot's most important data gathering is qualitative. Can all the necessary controls and switches be reached easily? What are the cockpit indications on production instruments of loss of control warning, departure, post-stall gyration, and spins? Can these indications be readily interpreted, or is the pilot so disoriented that he could not determine what action to take? What visual cues are available at critical stages of the recovery? Reference 10.14 gives an appropriate example of such a critical stage in the A-7D recovery sequence:

> On several occasions during recovery from fully developed spins, yaw rotation slowed, AOA decreased below 22 units, and roll rotation increased prior to release of anti-spin controls. Pilots found it easy to confuse roll rate for yaw rate leading to the "Auger" maneuver defined as rolling at unstalled AOA with anti-spin controls.

This sort of qualitative finding can be and usually is the most important kind of result from a spin test program. Hence, it is poor practice to ask the pilot to neglect cockpit observations to gather quantitative data which should be recorded by telemetry or on-board recording

devices. Project pilots must guard against this pilot overload by looking carefully at the available instrumentation, both airborne and ground-based.

10.4.4.2 FLIGHT TEST INSTRUMENTATION

The scope of the post-stall/spin test program will determine the extent of the instrumentation carried on board the aircraft. A qualitative program with a limited objective may require virtually no special instrumentation (10.16:1), while extensive instrumentation may be mandatory for a full-blown stall/post-stall/spin investigation. Table 10.6 shows typical flight test instrumentation required for post-stall/spin tests.

TABLE 10.6

TYPICAL FLIGHT TEST INSTRUMENTATION

Parameter	Time History[1]	Pilot's Panel
Angle of Attack	x	x
Production angle of attack	x	x
Angle of sideslip	x	
Swivel boom airspeed	x	x
Swivel boom altitude	x	x (coarse altimeter)
Production airspeed	x	
Production altitude	x	
Bank angle	x	

Pitch angle	x	
Pitch rate	x	
Roll rate	x	
Yaw rate	x	
Normal acceleration	x	x (sensitive indicator)
Accelerations at all crew stations	x	
All control surface positions	x	
Stick and rudder positions	x	
Stick and rudder forces	x	
All trim tab positions	x	
SAS input signals	x	
Engine(s) oil pressure	x	x
Hydraulic pressures	x	x
Fuel used (each tank)	x	x
Film, oscillograph, or tape-correlations and amount remaining	x	x
Event marker	x	x
Spin turn counter	x	x
Elapsed time	x	
Critical structural loads	x	x

Pilot warning signal (s)[2]		x
Emergency recover device indicators	x	x

[1] Magnetic tape, telemetry.

[2] Pilot warning signals may include maximum yaw rate indicators, spin direction indicators, minimum altitude indicators, and other such devices to help lower the pilot's workload. They may take the form of flashing lights, horns, oversized indicators, etc.

The prospective test pilot should particularly note the kinds of parameters to be displayed in the cockpit. In this area he must protect his own interests by assuring that the indicators and controls available to him are complete, but that they do not overload his capacity to observe and to safely recover the aircraft. Simulations, preferably under stress of some kind (in a centrifuge, for example), may help the pilot decide whether or not the cockpit displays and controls are adequate.

Finally, MIL-F-83691B, Paragraph 6.2.2.1.2.3, directs preparation of a technical briefing film and suggests that an aircrew training film be produced at the option of the procuring activity. Usually, it is advisable to have one or more movie cameras mounted on or in the test aircraft to provide portions of this photographic coverage. Motion pictures taken over the pilot's shoulder may provide visualization of the departure motion, readability of production instruments, information about the adequacy of the restraint system, or other similar data. A movie camera taking pictures of the control surface positions can produce dramatic evidence of the effectiveness or lack of effectiveness of recovery controls. These cameras and recording devices should be made as "crash-proof" or at least as "crash-recoverable" as possible.

Further information on flight test instrumentation, cockpit displays, and cameras may be found in Paragraphs 3.2.2, 3.2.3, and 3.2.4 of MIL-F-83691B.

10.4.4.3 SAFETY PRECAUTIONS

Stall/post-stall/spin test programs are usually regarded with suspicion by program managers and flying supervisors. Many such investigations have resulted in the loss of expensive, highly instrumented test aircraft and crew fatalities. Post-stall/spin tests are hazardous. Careful attention to detail in several areas will minimize the dangers involved.

10.4.4.3.1 Conservative Approaches. Use a conservative build-up approach to incrementally expand the areas of investigation, choosing safe increments until the aircraft's uncontrolled motions are better understood (10.4:3.4). How can the test pilot plan to assure that such an approach is actually followed?

First, the entire program is usually broken down into phases. Even the terms now in use - stall/post-stall/spin - suggest the basic phases of such an investigation, although in practice the phases are generally broken down in more detail. Tables I and II in MIL-F-83691B describe the recommended phases for such investigations.

Within these phases, there are several smaller steps to be taken with successive departures, post-stall gyrations, or spins. For example, aircraft loadings are normally changed gradually from clean to symmetric store loadings to asymmetric store loadings. The effects of these loading changes must be evaluated both for the aerodynamic effects and the changes in mass distribution. Unfortunately, it is not often obvious which effect is most volatile until after the tests are completed. One would also be ill-advised to use full pro-spin controls on the very first departure in phases B, C, or D. Delayed recoveries should be approached by sustaining the desired misapplication of controls in increments in each successive departure up to the maximum of 15 seconds as indicated in Phase D. Such conservatism in flying these tests is essential and must be adhered to scrupulously. However, it is also necessary to consider aircraft systems in order to plan a safe post-stall/spin program.

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10.4.4.3.2 Degraded Aircraft Systems. All systems are under an often unknown amount of strain during high angle of attack maneuvering. If the aircraft goes out of control in this flight regime, system design limits may well be exceeded. The propulsion/inlet system is often not designed to allow reliable operation of the engines during extreme angles of attack and sideslip. Engine flame out may result in loss of control in modern aircraft with hydraulic flight control systems. The test vehicle must have an alternate source of hydraulic power for the flight controls if there is a possibility that engine flameouts are likely to occur. However, do not overlook the behavior of the production hydraulic system: loss of production hydraulic pressure may be all that is necessary to prohibit intentional spins. In propeller-driven aircraft the hydraulic power used to govern the propeller pitch can also be a limiting factor, particularly during inverted spins. The electrical system may also be affected by engine flameout, and even a momentary failure can render instrumentation inoperative at a critical time. Hence, a reliable back-up electrical power source may be necessary. Other systems, such as the ejection system, pilot restraint system, or communications/ navigation system, may cause special problems during the post-stall/spin test program. The test pilot and test engineer must think through these special problems and, where necessary, add back-up systems to the test aircraft to assure safe completion of the program. Any backup systems that are required must not limit the range and scope of the tests; otherwise, they defeat their purpose.

10.4.4.3.3 Emergency Recovery Device. The ultimate back-up system, some sort of emergency recovery device, is so important that it deserves a paragraph all its own. Failure of this "last-ditch" system has in the past contributed to the discomfort of test pilot, engineer, and SPO director all too often. Reference 10.17 suggests that more attention must be given to the design of this system, perhaps to the extreme of making emergency recovery system components government-furnished equipment (GFE). While the feasibility of this rather drastic suggestion is questionable, it is imperative that more reliable systems be designed. Some of the things that must be scrutinized by the test pilot are:

1. Has the deployment/actuation mechanism demonstrated reliability through the expected envelope of dynamic pressures?

2. Are the moments generated large enough for all predicted spin rates?

3. Has the jettison mechanism demonstrated reliability throughout the expected envelope?

4. Are maintenance inspection procedures adequate for this system? (This system should be checked just prior to takeoff).

5. Does the emergency recovery system grossly alter the aerodynamic and/or inertia characteristics of the test aircraft?

Obviously, no such list is complete, but the test pilot must carefully evaluate every component of the emergency recovery system: spin chute, spin rockets, or any other device.

10.4.4.3.3.1 Spin-Recovery Parachute System Design (10.18). There are three distinctly different branches of technology involved in the design of a spin-recovery parachute system - parachutes, spinning, and airplane systems. For a given airplane, the spin-recovery parachute must be designed to recover the airplane from its worst spin condition. Definition of this "worst condition" and the parachute size and riser length is generally obtained for military airplanes from tests of dynamic models in the NASA Langley spin tunnel.

10.4.4.3.3.1.1 Parachute Requirements. Positive and reasonably quick opening (approximately three to four seconds) of the spin-recovery parachute is necessary for all operating conditions so that the spin may be terminated as rapidly as possible to minimize altitude loss.

A stable parachute is required so that it will tend to trail with the relative wind at the tail of the airplane in a spin and thus apply a yawing moment that is always anti-spin; whereas an unstable parachute because of its large oscillations may apply a yawing moment that varies from anti-spin to pro-spin, and thus hinders or prevents recovery.

Determination of the correct parachute size and riser length is very important in the overall design of a recovery system. The riser length controls the position of the parachute in the wake of the spinning airplane and therefore affects the force that the parachute can apply to the airplane. Figure 10.48 is an illustration of a typical spin-recovery parachute system.

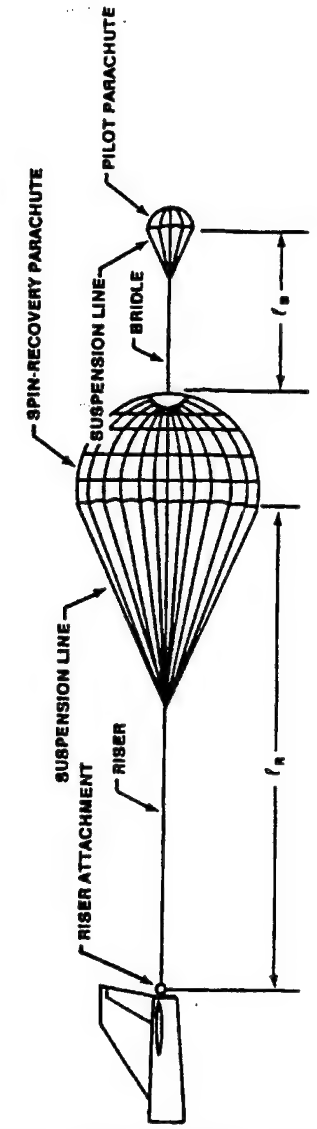


FIGURE 10.48. SKETCH OF SPIN-RECOVERY PARACHUTE SYSTEM AND ITS NOMENCLATURE(10.18)

10.4.4.3.3.1.2 Parachute Compartment (10.18). A fundamental requirement in any parachute installation is to locate the compartment and the riser attachment point as far aft on the airplane as possible. This approach will reduce the possibility of the riser or parachute striking the airplane and will also give the maximum moment arm for the parachute force to act on. It should be assumed that the angle the riser makes with the fuselage longitudinal axis can be as high as 90° if a flat or a highly oscillatory spin mode exists. If the riser is likely to contact the jet exhaust because of the attachment point location, then it must be protected against heat. Additional protection of the riser might be necessary if there is a possibility of its rubbing against the airplane structure after deployment. Since the riser generally is made of fabric (for example, nylon), abrasions on or nicks in the riser while it is in tension can cause it to fail very rapidly.

The parachute compartment should also be designed so that it does not change the spin and recovery characteristics of the airplane by changing the aerodynamic and/or inertia characteristics of the airplane with the installation and thereby invalidate the tests. Two types of parachute compartments are (1) one in which the compartment is permanently attached to the airplane and deployment is initiated by pulling the deployment bag from the compartment with a pilot parachute, and (2) one in which the compartment is pulled away and completely separated from the airplane by a pilot parachute which then pulls the compartment off the deployment bag when the riser is fully extended.

Two major requirements for a satisfactory parachute compartment are that it be designed so that (1) the extraction of the deployment bag by pilot parachute or tractor rocket or by forceful ejection can be accomplished regardless of the airplane attitude, and (2) the bag be undamaged during the deployment process.

10.4.4.3.3.1.3 Parachute Deployment Methods (10.18). The two basic methods for deploying the spin-recovery parachute from an airplane are the line-first and the canopy-first methods shown in Figure 10.49. The line-first method is preferred for several reasons, as indicated in the discussion of the method.

10.4.4.3.3.1.3.1 Line-first method. In line-first method (Figure 10.49a), a pilot parachute extracts the deployment bag from the parachute compartment, deploying first the riser, then the parachute suspension lines, and finally, the recovery parachute by pulling the deployment bag off the parachute. The primary advantage of this method is that it provides a clean separation of the deployment bag from the airplane and also ensures that the inflation of the spin-recovery parachute canopy will occur away from the airplane. Consequently, the possibility of the parachute fouling on the airplane and the effect of the airplane wake on the parachute are minimized. Furthermore, the snatch loads will be reduced because parachute inflation will occur after the riser is fully extended.

10.4.4.3.3.1.3.2 Canopy-first method. In the canopy-first method (Figure 10.49b), a pilot parachute extracts the deployment bag from the parachute compartment. A pilot parachute extracts the spin-recovery parachute canopy from the bag, then the suspension lines, and finally the riser. The primary disadvantages of this method are (1) the increased possibility of the spin- -recovery parachute canopy fouling on the airplane; (2) the high snatch loads that occur because the spin-recovery parachute canopy will become inflated before the riser has become fully extended; (3) the high opening shock loads; and (4) the possibility of the canopy being damaged, or only partly inflated, because the canopy and suspension lines become entangled. The only advantages of this method are (1) it requires a lower pilot parachute extraction force than the line-first concept because the spin-recovery parachute canopy is extracted easily regardless of the altitude of the spinning airplane; and (2) once the deployment starts, the parachute itself provides an additional force that helps complete the deployment of the canopy, suspension lines, and riser.

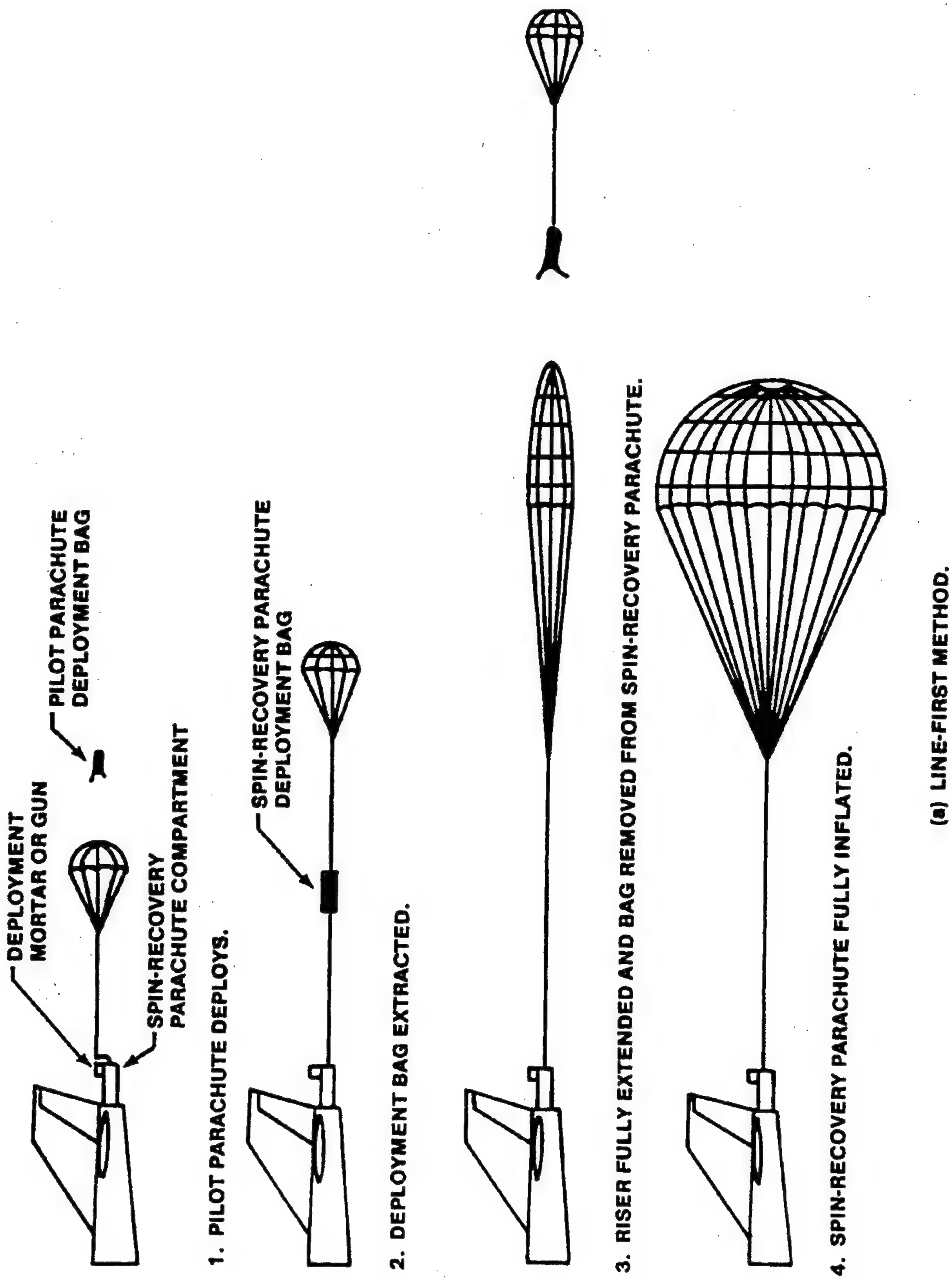


FIGURE 10.49. BASIC SPIN-RECOVERY PARACHUTE DEPLOYMENT TECHNIQUE

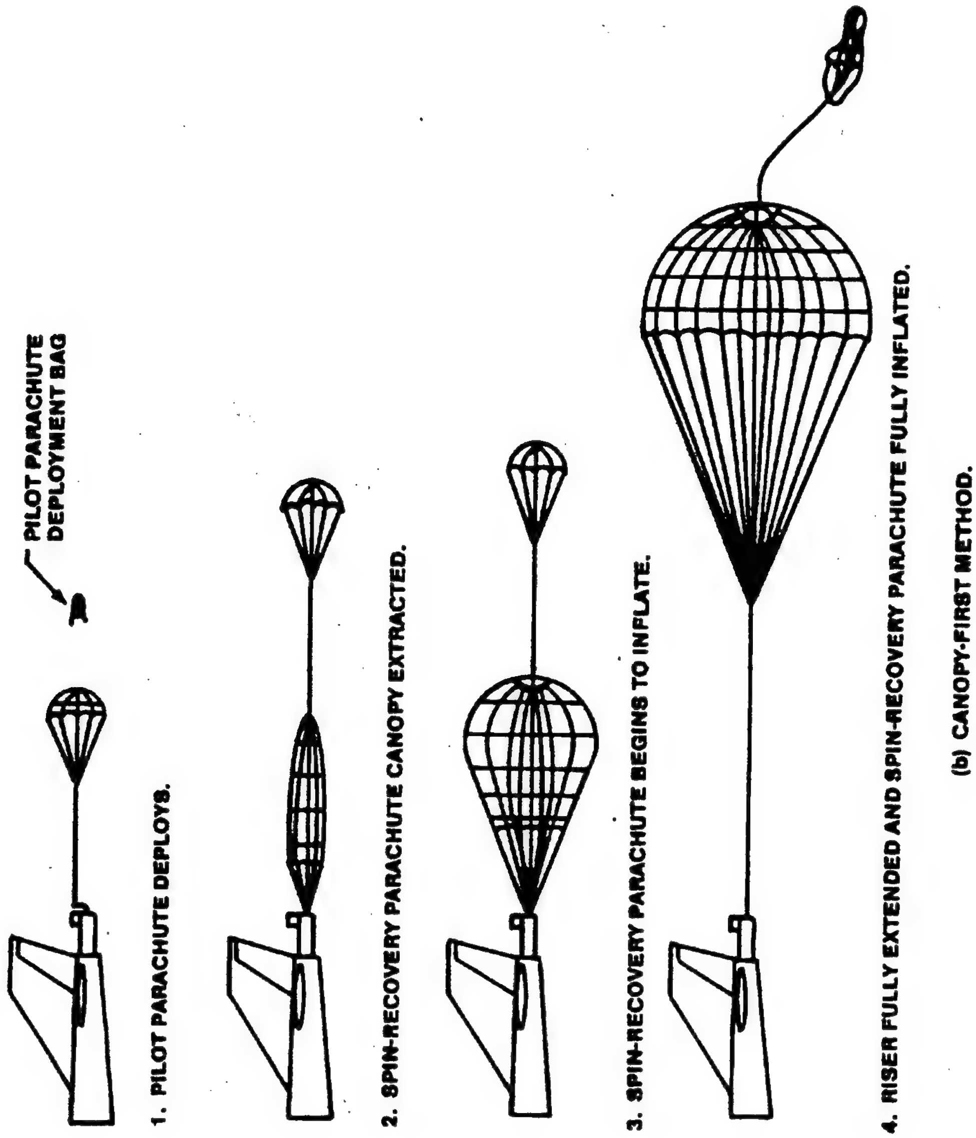


FIGURE 10.49. CONCLUDED

10.4.4.3.3.1.4 Basic Attachment Methods (10.18). The spin-recovery parachute riser is attached to the airplane by an attachment and release mechanism and this device has proven to be a critical item in the system design. For this reason, regardless of the type of mechanism used, no part of it should require such precise adjustment that lack of such adjustment could cause the mechanism to malfunction. The mechanism must perform the following critical functions: (1) attachment of the parachute riser to the airplane, (2) release of the parachute after spin recovery, and (3) automatic release of the parachute in the event of inadvertent deployment during critical phases of flight. There are two basic methods normally used:

1. Closed-jaw method (Figure 10.50a) - The attachment of the riser to the airplane is made prior to take-off and provision is made for automatic release in the event of premature deployment.

2. Open-jaw method (Figure 10.50b) - The attachment is not made until immediately before a spin test.

Several factors must be considered in designing the attachment and release mechanism. For example, if the shackle, or D-ring, is locked in the attachment mechanism prior to take-off, as illustrated by the closed-jaw concept of Figure 10.50, it is essential from the standpoint of flight safety that provision be made so that the parachute will automatically jettison should it inflate inadvertently. This automatic jettisoning of the parachute can be accomplished by putting a weak link, such as a shear pin, in the system. Prior to the start of the spin tests, the weak link is bypassed by a locking mechanism capable of withstanding the opening shock load of the parachute. If the mechanism is left open until the start of the spin tests, however, as illustrated by the open-jaw concept of Figure 10.50b, the para- chute would be automatically jettisoned since it would be unrestrained.

This approach does require, however, that steps be taken to ensure that the shackle is in position in the mechanism when the time comes to arm the system. A low-strength bolt or safety wire can be used to achieve the proper positioning. For either of the foregoing types of systems, a light is generally used to indicate that the system has been armed by bypassing the weak link or by closing the jaws.

In both the closed-jaw and open-jaw methods, the normal procedure for releasing the parachute after it has been deployed is by mechanical means. Provisions, however, should be made for emergency jettisoning of the parachute if the primary jettison system fails to operate. This jettisoning can be accomplished through the use of explosive bolts or pyrotechnic line cutters. If the explosive bolts are used, they should be of the nonfragmenting variety to ensure the safety of the airplane. The pyrotechnic line cutters have a disadvantage in that the cutters and the electric wires to them are subject to damage by the slipstream and therefore might fail to function.

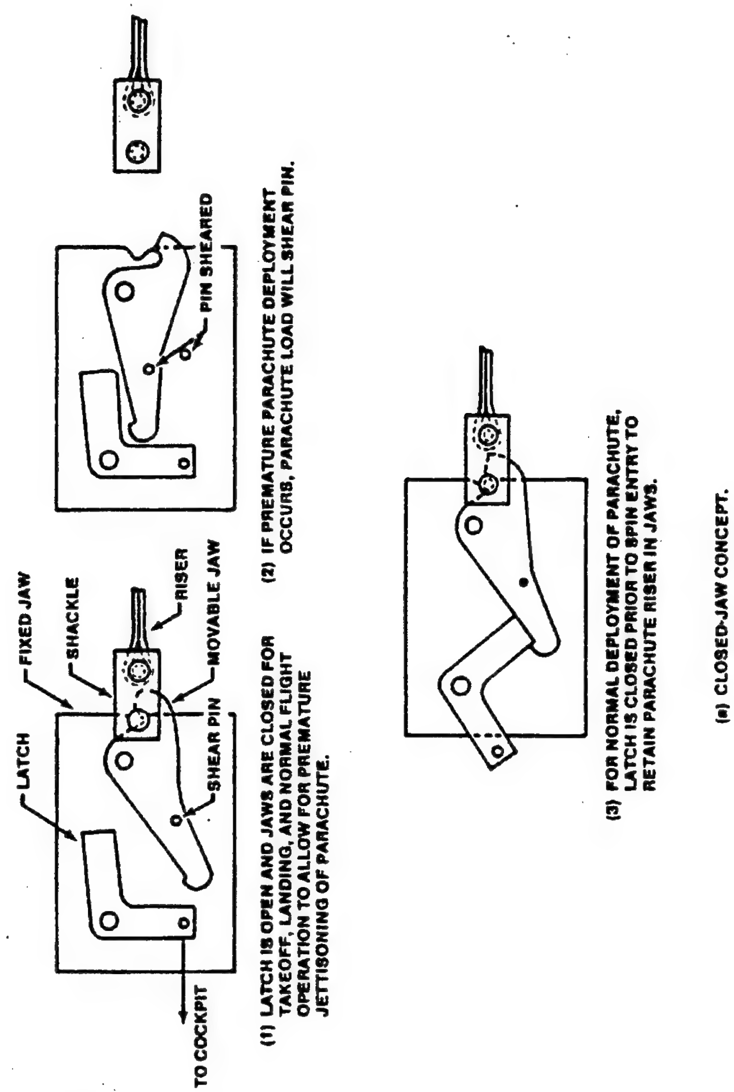


FIGURE 10.50. PARACHUTE ATTACHMENT AND RELEASE MECHANISMS

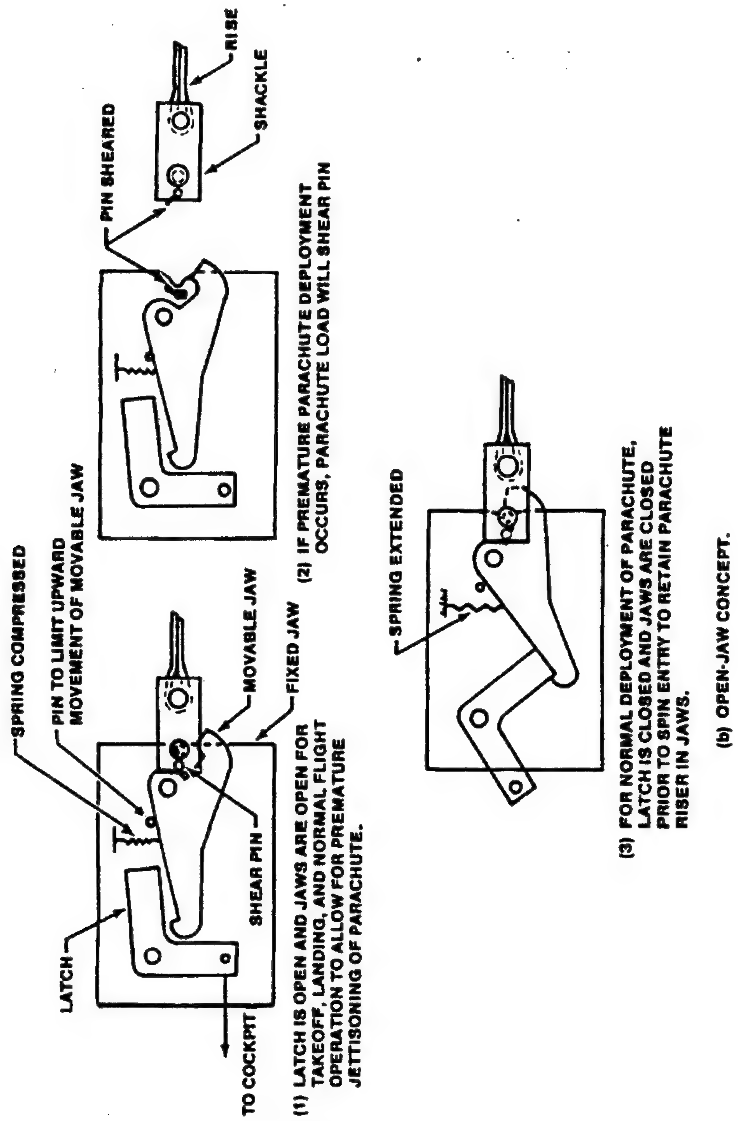


FIGURE 10.50. CONCLUDED

10.4.4.3.3.2 Alternate Spin-Recovery Devices. Although tail-mounted spin-recovery parachutes are used almost exclusively in full-scale spin demonstrations, rockets and wing-tip-mounted parachutes have been considered. Anti-spin rockets have been used occasionally, but wing-tip mounted parachutes have been used apparently only once.

10.4.4.3.3.2.1 Rockets (10.18). Rockets are generally used for spin recovery in special cases where the use of a parachute involves unusual problems. Tail-boom airplane configurations or tailless configurations with a very short tail moment arm might provide such unusual problems. Rockets, however, have many disadvantages when compared with tail-mounted parachutes as will be discussed later.

Rockets generally have been installed on or near each wing tip but there have been cases where the rocket was installed at the nose or the tail of the airplane. When the rockets are installed on the wing tips, their thrust is applied in a forward direction. Depending on the direction of the spin which should be determined by a sensor, the left or right rocket is fired to apply an anti-spin yawing moment (for example, in a right spin the right rocket would be fired).

When the spin-recovery rockets are added to the airplane, care should be taken that the rocket installation does not alter the spin and recovery characteristics of the airplane by altering the aerodynamic and/or inertia characteristics of the airplane and thereby invalidate the tests.

10.4.4.3.3.2.1.1 Thrust orientation. The effectiveness of the applied yawing moment produced by rockets mounted on the wing tips depends on the orientation of the rocket thrust line with respect to the principal axis of the airplane. In order to avoid a rolling moment that might be adverse, depending on the mass distribution of the airplane the rocket thrust should be aligned as closely as possible with the principal axis of the airplane.

10.4.4.3.3.2.1.2 Rocket impulse. On the basis of past experience with model spin-recovery rocket investigations, certain conclusions can be drawn regarding the nature of the rocket impulse required, rocket impulse being the product of the average value of the thrust and the time during which it acts. The rocket must not only provide a sufficient yawing moment for recovery, but the rocket must provide this moment for as long as the spin rotation is present. Rockets that have the same impulse but different amounts of thrust and thrust durations may or may not produce satisfactory spin recoveries depending on the magnitude of the thrust and the thrust duration.

The primary advantages of a rocket-recovery system are:

(1) Definite known yawing moment is applied.

(2) Applied yawing moment is not affected by wake of airplane.

(3) Rockets do not have to jettisoned after use.

(4) Fuselage or wing has to be strengthened only to withstand the yawing moment produced by the rockets.

The disadvantages of rockets are:

(1) Some type of sensor must be used to determine the diretion of spin so that the proper rocket is fired.

(2) Duration of rocket thrust is limited.

(3) If duration of rocket thrust is too long and pilot does not terminate it when recovery is complete, the airplane may enter a spin in the opposite direction; conversely, if the rocket thrust is terminated prematurely the airplane may not recover from the spin.

(4) If the pilot does not regain control of the airplane following recovery by use of a rocket and the airplane enters a second spin there is no further emergency recovery system; whereas, with tail-mounted recovery parachute, he can retain the stabilizing effect of the parachute until he is sure he has rocovered control.

(5) Two installations are necessary if rockets are mounted on wing tips.

10.4.4.3.3.3 Wing-Tip-Mounted Parachutes (10.18). Tests were conducted until 1952 in the NASA Langley spin tunnel on dynamically scaled models using wing-tip-mounted parachutes. Full-scale airplane tests with wing-tip parachutes have apparently been made on only one airplane in the past 20 years. Wing-tip parachutes apply an anti-spin yawing moment to the airplane to effect a spin recovery; they also apply a rolling moment and, if the airplane has a swept wing, a pitching moment will be applied. Even though wing-tip parachutes generally need be only about 50 to 60 percent as large as a tail parachute in order to effect a spin recovery, they have all the disadvantages of rockets. In addition if the mass of the air-

plane is distributed along the wing, the rolling moment produced by the parachute will retard spin recoveries.

10.4.4.3.4 Special Post-Stall/Spin Test Flying Techniques. In general, the test pilot must have indelibly fixed in mind what control actions he will take when the first departure occurs. An inadvertent departure can give just as meaningful (perhaps more meaningful) data as an intentional one - if the test pilot overcomes his surprise quickly enough to make preplanned and precise control inputs. The keys to avoiding confusion in the cockpit have already been mentioned, but they bear repeating. The test pilot must be recently proficient in post-stall gyrations and in spinning, and he must be so familiar with the desired recovery controls that they are second nature. Apart from overcoming the surprise factor through adequate preparation, the test pilot may need some other tricks in this highly specialized trade.

10.4.4.3.4.1 Entry Techniques

10.4.4.3.4.1.1 Upright Entries. For aircraft susceptible or extremely susceptible to spins, an upright spin may be easy to attain. In this case the test pilot's main concern may be how to produce repeatable characteristics; that is, he may seek to achieve the same entry g-loading, attitude, airspeed, and altitude in successive spins so that correlation between spins is easier. Of course, if the aircraft is resistant to spins, it may still be susceptible to departure and entry into a post-stall gyration. In this case, correlation of the data may be even more difficult since the random motions of a PSG are seldom repeatable. Again, the attempt usually is to achieve repeatable entry conditions so that over a large statistical sample the characteristics of the PSG become clear. Achieving several departures with repeatable entry conditions is one of the more demanding piloting tasks. Considerable proficiency is required to achieve the AOA bleed rates or airspeed bleed rates specified in MIL-F-83691B. Once the baseline characteristics for a given configuration are relatively well known, the test pilot is called on to simulate entries appropriate to the operational use of the aircraft.

10.4.4.3.4.1.2 Tactical Entries. These entry maneuvers must be carefully thought out in light of the expected role of the aircraft. It is often wise to consult directly with the using command, particularly if the aircraft has already entered operational service. Reference 10.4 suggests the types of tactical entries listed in Table 10.8 but past experience is no substitute for foresight in planning such tests. By carefully examining the tactics envisioned by operational planners, the test pilot should be able to recognize other possible tactical entries which may cause difficulty in the high angle of attack flight regime.

TABLE 10.8. TACTICAL ENTRIES

1. Normal inverted stalls
2. Aborted maneuvers in the vertical plane (vertical reversals, loops, or Immelmans)
3. High pitch attitudes (above 45°)
4. Hard turns and breaks as used in air combat maneuvering
5. Overshot roll-ins as for ground attack maneuvering
6. High-g supersonic turns and/or transonic accelerations/decelerations
7. Sudden idle power and/or speed brake decelerations
8. Sudden asymmetric thrust transients prior to stall

10.4.4.3.4.1.3 Inverted Entries. Obtaining entries into inverted post-stall gyrations or spins can be very difficult simply because aircraft often lack the longitudinal control authority to achieve a stall at negative angles of attack. The most straightforward way to depart the aircraft in an inverted attitude is to roll inverted and push forward on the stick until stall occurs at the desired g-loading. Many aircraft, however, have marginal elevator authority and it is necessary to misapply the controls to obtain an inverted departure. Pulsing the rudder or applying other pro-spin controls as the nose drops can help precipitate departure. In the OV-10, for example, the direction of applied aileron determines the direction of the inverted spin - provided full aileron deflection is used. However, if aerodynamic

controls lack authority, the test pilot can also use inertial moments to precipitate inverted departures.

How the inertial terms can aid entry into a spin can best be seen by examining the pitch rate acceleration equation:

$$\dot{q} = \frac{-M_a}{I_y} + p\,r\left(\frac{I_z - I_x}{I_y}\right)$$

If the negative pitching acceleration generated by M_{aero}/I_y was too small to produce a stalled negative angle of attack, an additional negative pitching acceleration can be produced from (pr $(I_z - I_x)/I_y$). All that is necessary is for p and r to have opposite signs. Typically, the roll momentum is built up by rolling for at least 180° opposite to the desired direction of the inverted spin and then applying full prospin controls at the inverted position. Obviously, these control manipulations must be made at an angle of attack near the stall. Sometimes it is even advisable to apply a slight amount of rudder opposite to the roll during the roll momentum buildup period. A typical procedure designed to produce a left inverted spin is given below:

1. Establish a nose high pitch attitude.

2. Apply full right aileron and a slight amount of left rudder.

3. After a minimum of 180° of roll (360° or more may be advantageous in some aircraft), apply full left rudder, maintain full right aileron, and full forward stick (on some aircraft full aft stick may be used).

4. Recover using predicted or recommened recovery procedures.

This procedure must be modified to fit the characteristics of a particular aircraft, but it does illustrate the kind of control manipulation sometimes required in post-stall/spin investigations. Some aircraft will not enter an inverted spin using this sort of exaggerated technique, but using the inertial moments to augment aerodynamic controls has uncovered spin modes not

obtained by other means. Reference 10.12 provides further information on the subject of inverted spinning.

10.4.4.3.4.2 Recovery Techniques

Out-of-Control Recoveries

The underlying principle of all recovery techniques is simplicity (refer to Paragraph 3.4.2 of MIL-F-83691B). The procedure to be used must not require the pilot to determine the nature or direction or the post-stall gyration. In fact, Paragraph 4.8.4.3.2 of MIL-STD 1797A requires recovery from both post-stall gyrations and incipient spins using only the elevator control. Engine deceleration effects must be tested. Any part of the flight control system (the SAS, for example) which hinders desired control surface placement must be identified and carefully evaluated. Care must be taken to ensure that the recovery controls recommended to recover from a post-stall gyration will not precipitate a spin. The test pilot is primarily responsible for identifying reliable visual and cockpit cues to distinguish between post-recovery angular motions (steep spirals, rolling dives) and the post-stall gyration. Taken together, these requirements demand that the test pilot be a careful observer of the motion. In fact, he is likely to become so adept at making these observations that he must guard against complacency. His familiarity with the motions may cause him to over-estimate the operational pilot's ability to cope with the out-of-control motions. Paragraph 4.8.4.3.2 of MIL-STD 1797A specifies that the start of the recovery shall be apparent to the pilot within three seconds after initiation of recovery. This requirement is very stringent and will require very fine judgement on the part of the test pilot.

10.4.4.3.4.2.1 Spin Recoveries. The criteria for recovery from a spin are outlined in Table 10.9. (Paragraph 4.8.4.3.2 of MIL-STD 1797A). These criteria are applicable to any spin modes resulting from any control misapplication specified in MIL-F-83691B. Timing of control movements should not be critical to avoid spin reversals or an adverse mode change.

Table 10.10 outlines the NASA Standard, NASA Modified, and NASA Neutral recovery procedures. These recoveries are by no means optimum for all aircraft and they must not be

construed to be. In contrast, the F-4E recovery technique includes forward stick, which reflects the philosophy of simplifying out-of-control recovery procedures. Generally, forward stick is desirable for recovery immediately following a departure. The reason for retaining the forward stick is to keep the out-of-control recovery procedure like the spin recovery procedure. However, individual aircraft characteristics may dictate that out-of-control recovery procedures differ from spin recovery procedures. Such characteristics violate the specifications of both MIL-STD 1797A and MIL-F-83691B, but the test pilot must evaluate the need for two recovery procedures. He cannot assume that any "canned" recovery procedure will work nor that the design meets the specifications. In summary, the test pilot's job is to assure that the operational pilot has a simple, reliable recovery procedure which will consistently regain controlled flight.

TABLE 10.9. RECOVERY CRITERIA

Class	Flight Phase	Turns for Recovery
I	Category A,B	1-1/2
I	PA	1
IV	Category A,B	2-1/2

TABLE 10.10. RECOVERY TECHNIQUES

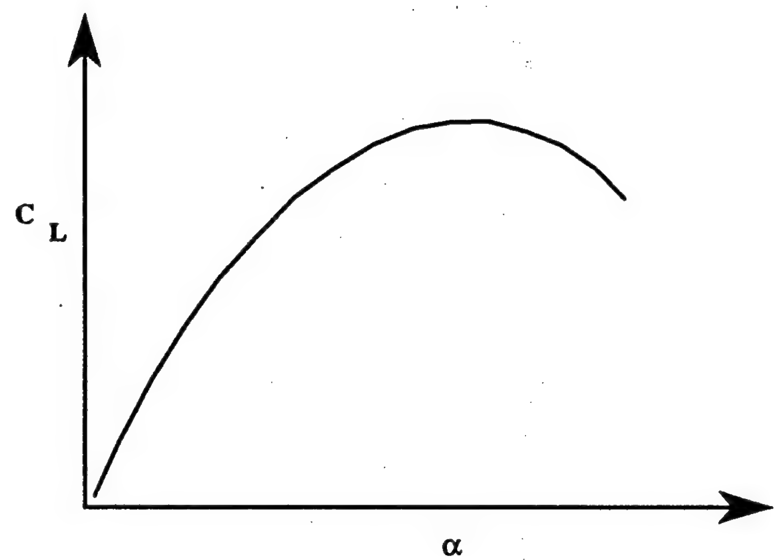
NASA Standard	NASA Modified	NASA Neutral
(If ailerons were held during spin, neutralize)	Same	Neutralize all Controls
A. Full opposite rudder	A. Full opposite rudder and at the same time ease stick forward to neutral.	
B. Stick full aft	B. Neutralize rudder when rotation stops.	
C. When rotation stops - neutralize rudder (immediately)		
D. <u>EASE</u> stick forward to approximately neutral position.		

10.5 PROBLEMS

10.1 List and describe two factors that reduce the kinetic energy in the boundary layer and lead to flow separation.

10.2 Given the following C_L vs α curve, show the effect of:

a. adding boundary layer control,
b. increasing aspect ratio and
c. adding flaps



10.3 Considering three-dimensional stall effects, give three reasons contributing to the pitch up tendencies of swept-wing, T-tail aircraft (such as the F-101).

10.4 If the 1G stall speed of a 20,000 pound F-16 is 110 KEAS, what is the stall speed at 9Gs with an additional 5,000 pounds of stores.

10.5 For an aircraft that stalls at 120 knots in the power approach configuration, what is the MIL-SPEC acceptable airspeed range for the onset of stall warning?

10.6 Which is the preferred MIL-SPEC solution to providing adequate stall warning?

a. Appropriate aerodynamic design and mass distribution.
b. Special devises (such as stall horns and stick shakers).
c. Intolerable buffet between 82% and 90% $C_{L\ stall}$.

10.7 Does the MIL-SPEC allow the use of throttles to assist in stall recovery?

10.8 Define post stall gyration.

10.9 Describe a Phase C Stall.

10.10 What is the difference between an incipient and a developed spin?

10.11 Complete the table:

SPIN MODE MODIFIERS

Sense	Attitude	Rate	Oscillations

10.12 When does the directional departure parameter $Cn_{\beta dyn}$ predict a departure?

10.13 Using diagrams, show why a stalled wing with an initial yaw rate generates an autorotative couple.

10.14 Using a diagram, show how an oval nose shape affects a spinning aircraft. (Is it pro-spin or anti-spin)?

10.15 List I_x, I_y and I_z in decreasing order of magnitude for:

a. F-104

b. YB-49 (Flying Wing)

10.16 List basic assumptions for a fully developed spin.

10.17 Complete the following equations with an =, <, or > sign so they satisfy prerequisites for a fully developed, upright spin.

a. C_l 0

b. Cn 0

c. $C_{m,b}$ $-M_{aero}$

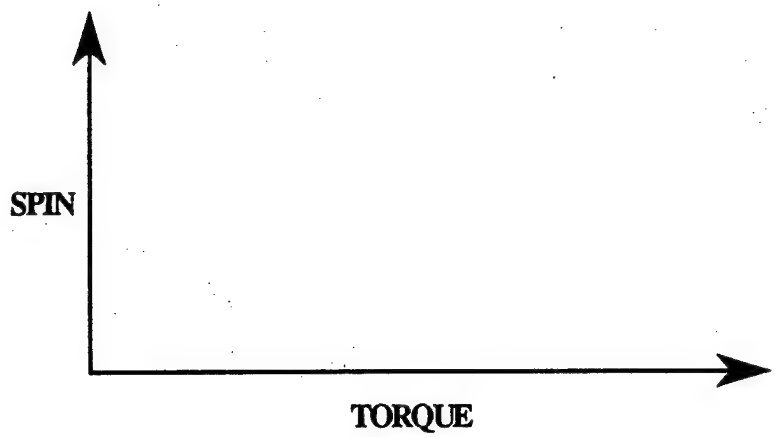
d. $\partial C_{m,b}/\partial\alpha$ 0

e. α α_{stall}

f. $\partial C_l/\partial\omega$ 0

g. $\partial Cn/\partial\omega$ 0

10.18 Given the following spin and torque axes, show the precession axis.



10.19 When viewed from the rear, the A-37 engines spin clockwise. As the A-37 nose pitches down at the stall, what direction will the gyroscopic forces of the engines make the aircraft spin?

10.20 List two ways that a fuselage-loaded aircraft spins differently than a wing-loaded aircraft.

10.21 In an inverted spin, is the spin direction determined by the sense of the yaw rate or the sense of the roll rate?

10.22 What cockpit instrument indicator is the most useful for differentiating between an upright and an inverted spin? For determining the yaw direction?

10.23 Define "recovery."

10.24 List four general categories of spin recovery methods.

10.25 Given a fuselage loaded, erect aircraft spinning to the right, how well ailerons applied to the left affect the spin? Support your answer with diagrams.

10.26 Given a wing loaded, inverted aircraft spinning to the left, how will ailerons applied to the left affect the spin? Support your answer with diagrams.

10.6 ANSWERS

10.1 List and describe two factors that reduce the kinetic energy in the boundary layer and lead to flow separation.

a. Viscous friction: Energy loss varies with surface roughness and distance traveled.

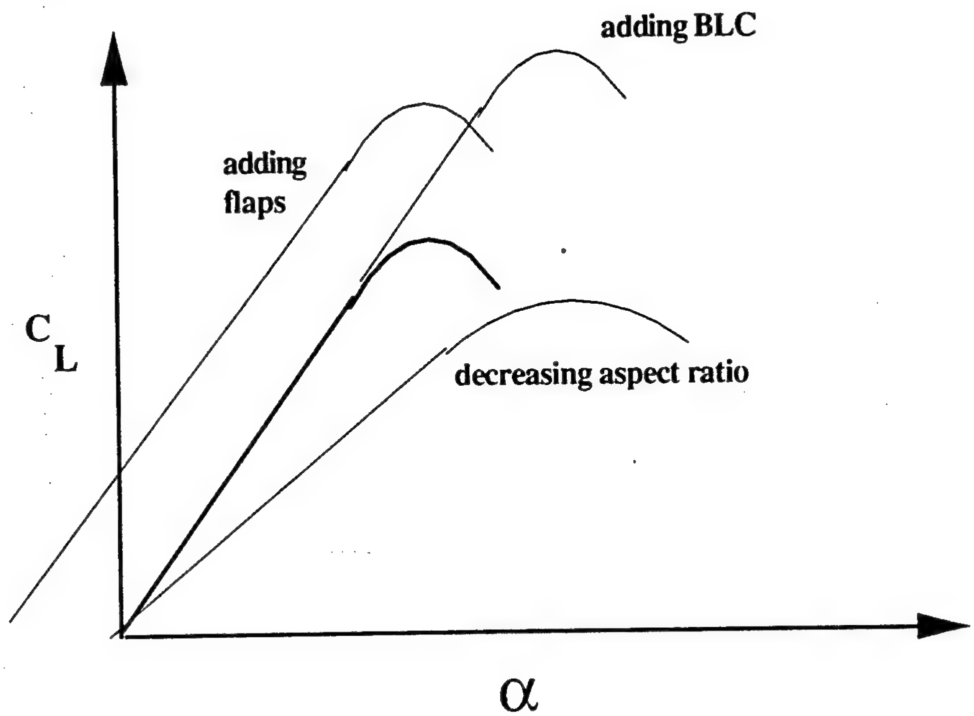
b. Adverse pressure gradient: Boundary layer energy is dissipated as the air moves against the adverse pressure gradient above a cambered airfoil section. The rate of energy loss is a function of.

1. Body contours - Camber, thickness distribution, and sharp leading edges are examples.

2. Angle of attack - Increased angle of attack steepens the adverse pressure gradient.

10.2 Given the following C_L vs α curve, show the effect of:

a. adding boundary layer control,
b. increasing aspect ratio and
c. adding flaps



10.3 Considering three-dimensional stall effects, give three reasons contributing to the pitch up tendencies of swept-wing, T-tail aircraft (such as the F-101).

a. Wing root section remains unstalled proding greater lift
b. In board movement of wingtip vortex system produces greater downwash behind the center of the wing
c. Increaased downwash produces greater download by horizontal tail causing uncontrollable pitch up

10.4 If the 1G stall speed of a 20,000 pound F-16 is 110 KEAS, what is the stall speed at 9Gs with an additional 5,000 pounds of stores.

$$L = n\,W$$

at stall $\quad L = C_{L_{max}}\, 1/2\ \rho V^2_{min} S$

assuming $C_{L_{max}}$, ρ, S are constant: $\quad L = K\, V^2_{min}$

$$L = (1)\,(20{,}000) = K\,(110)^2$$
$$K = 1.65$$

Solving for the new conditions:

$$L = (9)\,(25{,}000) = 1.65\ V^2_{min}$$
$$V_{min} = 369 \text{ KEAS}$$

10.5 For an aircraft that stalls at 120 knots in the power approach configuration, what is the MIL-SPEC acceptable airspeed range for the onset of stall warning?

Minimum is higher of 1.05 Vs or Vs + 5 : 126 or 125
Maximum is higher of 1.10 Vs or Vs + 10 : 132 or 130

Acceptable range is 126 - 132 knots

10.6 Which is the preferred MIL-SPEC solution to providing adequate stall warning?

a. Appropriate aerodynamic design and mass distribution.
b. Special devises (such as stall horns and stick shakers).
c. Intolerable buffet between 82% and 90% $C_{L\ stall}$.

10.7 Does the MIL-SPEC allow the use of throttles to assist in stall recovery?

No, unless leaving the throttles fixed would exceed engine operating limitations.

10.8 Define post stall gyration.

Uncontrolled motion about one or more airplane axes occuring in the post-stall flight regime. While this type of airplane motion involves angles of attack higher than stall angle, lower angles may be encountered intermittently.

10.9 Describe a Phase C Stall.

Stalls with aggravated and sustained control inputs.

10.10 What is the difference between an incipient and a developed spin?

The incipient spin is the initial, transitory phase of the spin during which it is not possible to identify the spin mode. The developed spin is the phase of the spin during which it is possible to identify the spin mode.

10.11 Complete the table:

SPIN MODE MODIFIERS

Sense	Attitude	Rate	Oscillations
Upright	Extremely Steep	Slow	Smooth
Inverted	Steep	Fast	Mildly Oscillatory
	Flat	Extremely Fast	Oscillatory
			Highly Oscillatory
			Violently Oscillatory

10.12 When does the directional departure parameter $Cn_{\beta dyn}$ predict a departure?

When $C_{n\beta}$ is negative, the airplane will continue to yaw away form the β (statically unstable)

10.13 Using diagrams, show why a stalled wing with an initial yaw rate generates an autorotative couple.

See Figures 10.15 - 10.19

10.14 Using a diagram, show how an oval nose shape affects a spinning aircraft. (Is it pro-spin or anti-spin)? Pro-spin

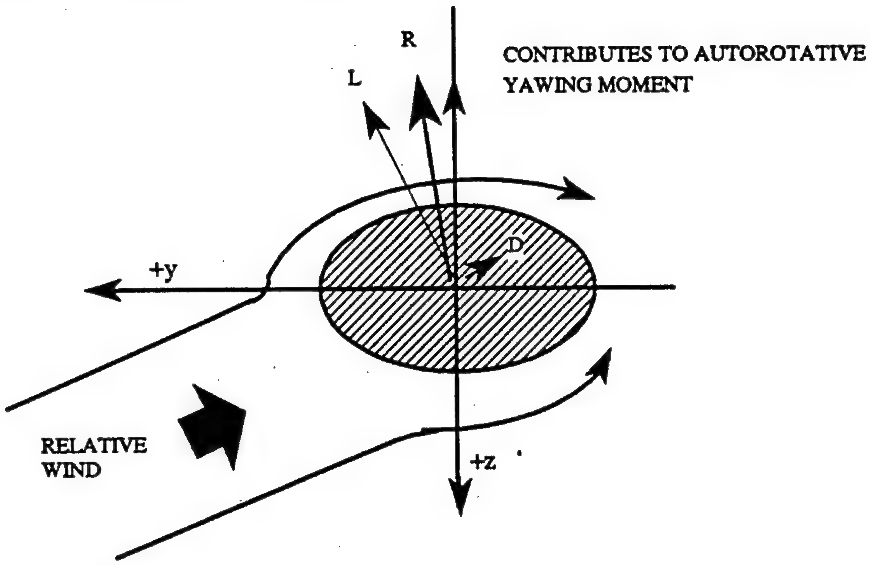


FIGURE 10.20A. PLAIN FUSELAGE

10.15 List I_x, I_y and I_z in decreasing order of magnitude for:

a. F-104

$I_z > I_y > I_x$

b. YB-49 (Flying Wing)

$I_z > I_x > I_y$

10.16 List basic assumptions for a fully developed spin.

a. $M_i + M_a = 0$
b. $\dot{p} = \dot{q} = \dot{r} = \omega = v = 0$
c. $q = 0$
d. $\beta = 0$
e. $C_m < 0$
f. $C_{m\alpha} < 0$
g. $\alpha > \alpha_S$
h. Sustained Yaw Rate

10.17 Complete the following equations with an =, <, or > sign so they satisfy prerequisites for a fully developed, upright spin.

a. $C_l = 0$

b. $Cn = 0$

c. $M_{inertial} = -M_{aero}$

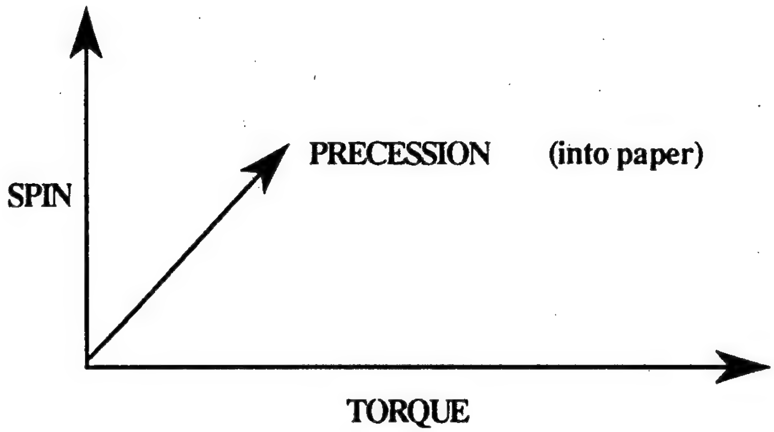
d. $\partial C_{m,b} / \partial \alpha < 0$

e. $\alpha > \alpha_{stall}$

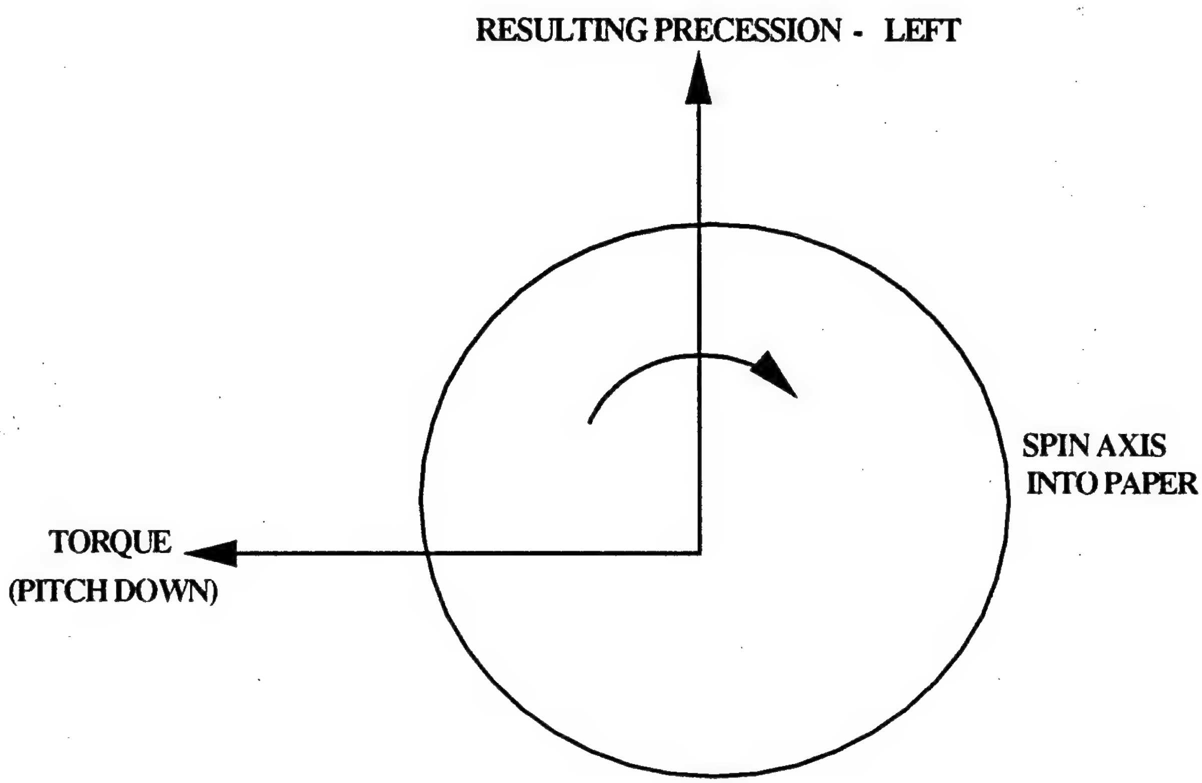
f. $\partial C_l / \partial \omega < 0$

g. $\partial Cn / \partial \omega < 0$

10.18 Given the following spin and torque axes, show the precession axis.



10.19 When viewed from the rear, the A-37 engines spin clockwise. As the A-37 nose pitches down at the stall, what direction will the gyroscopic forces of the engines make the aircraft spin?



AS VIEWED FROM REAR

10.20 List two ways that a fuselage-loaded aircraft spins differently than a wing-loaded aircraft.

More Oscillatory and Flatter

10.21 In an inverted spin, is the spin direction determined by the sense of the yaw rate or the sense of the roll rate?

Always the sense of yaw rate in any spin

10.22 What cockpit instrument indicator is the most useful for differentiating between an upright and an inverted spin? For determining the yaw direction?

AOA differentiates upright or inverted spin. The turn needle shows yaw direction.

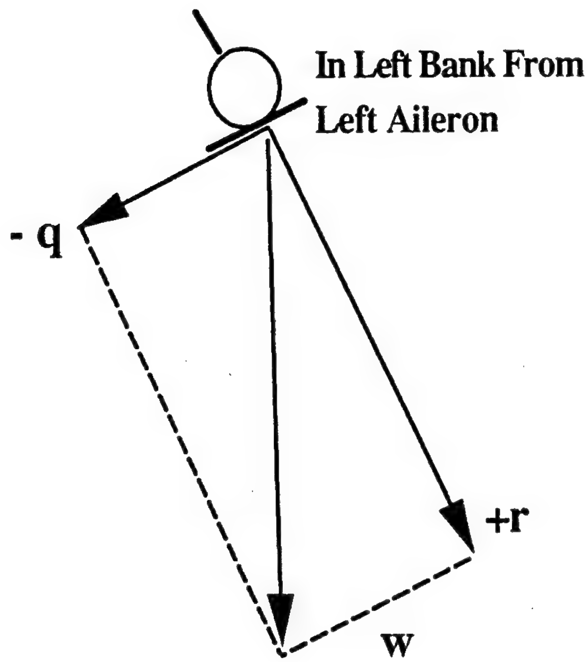
10.23 Define "recovery."

The transitional event from out of control conditions to controlled flight.

10.24 List four general categories of spin recovery methods.

a. Modify aerodynamic moments with flight controls or configuration changes
b. Reposition the aircraft attitude with respect to the spin axis
c. Vary engine power
d. Add spin chutes, rockets, etc.

10.25 Given a fuselage loaded, erect aircraft spinning to the right, how well ailerons applied to the left affect the spin? Support your answer with diagrams.



$$\dot{r} = \ldots + \frac{I_x - I_y}{I_z}\, p\, q + \ldots$$

erect right spin r and P are +

fuselage loaded $I_x - I_y$ is ---

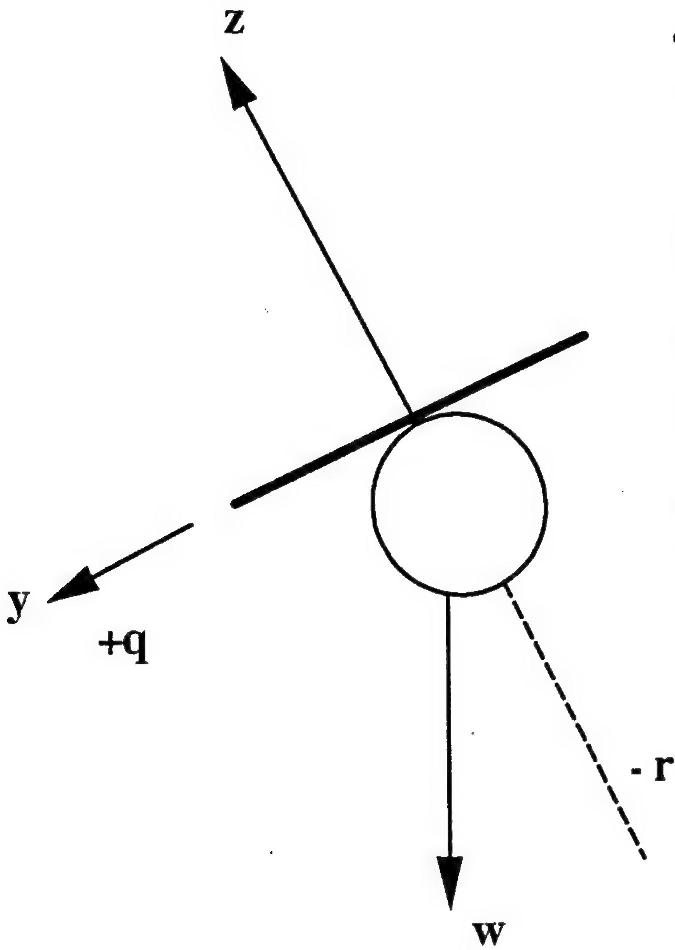
applying left aileron generates - q

$\dot{r} = (-)\ (+)\ (-) = +$

$\dot{r}$ is + and r is +

so left aileron is a pro spin input

10.26 Given a wing loaded, inverted aircraft spinning to the left, how will ailerons applied to the left affect the spin? Support your answer with diagrams.



$$\dot{r} = \dots + \frac{I_x - I_y}{I_z} \, p \, q + \dots$$

inverted left spin r is --- and p is + wing loaded $I_x - I_y$ is +

applying left ailerons generates +q

$\dot{r}$ = (+) (+) (+)

$\dot{r}$ = + and r is —

so left aileron is an anti spin input

Extract from AFFTC F-5F Spin Report

Engine Operating Characteristics

Four double-engine and five single-engine flameouts were experienced during 195 maneuvers conducted during this program. The susceptibility of the engines to flameouts was dependent upon power setting and severity of the post-stall motions. Most of the flameouts occurred at power steerings at or above 95% RPM and above 40° AOA with large sideslip angles present. The flameouts typically occurred during the first significant AOA/sideslip excursion during the post-stall motions. The probability of engine flameout was significantly reduced when power during the maneuver entry was reduced to below 95% RPM. Left and right engines were equally susceptible to flameout during a given maneuver. All of the flameouts occurred in the vicinity of 35,000 feet pressure altitude. Flameout susceptibility should be less at lower altitudes. No engine flameouts were encountered during inverted out-of-control flight. These characteristics are the same as those obtained on the F-5E.

Aerodynamic Analysis

This section contains an explanation of the aerodynamics involved in the airplane characteristics which were described in previous sections. Trends in airplane stability and control were derived from analysis of flight test data. Similar trends are generally substantiated in wind tunnel and analytical data. Wind tunnel data presented in Appendix B was obtained as low Mach but exhibits trends evidenced throughout the subsonic Mach range.

Basic Airplane Erect Characteristics

Longitudinal

The slight nose drop tendency at stall (21 to 25 degrees AOA, depending on Mach) resulted from a decrease in lift and a significant increase in static longitudinal stability above stall AOA. The decrease in lift, as indicated by a local change in sign of the slope of normal force coefficient (C_n) versus AOA, is presented in Figures B1 and B2. The increase in

longitudinal stability, as indicated by an increase in slope of pitching moment (C_m) versus AOA, is presented in Figures B3 and B4. Normal force coefficient again increased when AOA exceeded approximately 25°, as indicated by a restoration in the slope of C_n versus AOA, but was not always apparent to the pilot. Although C_n again increased above 25° AOA, a decrease in speed (due to increased drag) usually meant a decrease in normal load factor as AOA increased.

Flight test data indicated static longitudinal stability (C_{m_α} at zero sideslip) as being stable up to 40° AOA, essentially neutral between 40 and 50° AOA, and again stable above 50° AOA. For post-stall AOA's less than 40°, longitudinal stability was only slightly reduced over that of the F-5E for respective nominal and aft cg's (nominal cg of the F-5E is 14% MAC and aft cg is 20% MAC). Above 40° AOA, the F-5F exhibited significantly less longitudinal stability than the F-5E. Above 50° AOA with an aft cg (16% MAC), the F-5F had approximately 40% less nosedown restoring pitching moment than the F-5E which resulted in the decreased PSG and spin resistance of the F-5F (the other difference being large yawing moments at small sideslip angles, discussed later). Trim AOA's with full aft stick were 28 and 31° with nominal and aft cg's,respectively, but higher AOA's were obtained when full aft stick was sustained. Directional instabilities and strong dihedral effect resulted in wing rock with considerable oscillations in roll rate, yaw rate, and sideslip when full aft stick was sustained. Inertial pitch coupling (I_z - I_x/I_y pr) tended to increase AOA above maximum trim AOA. Also, a noseup pitching moment due to sideslip (positive C_{m_β}) existed above approximately 28° AOA. When full aft stick was sustained, sideslip oscillations reduced or eliminated the nosedown aerodynamic pitching moment, or even resulted in a net noseup moment, depending on the magnitude of the sideslip oscillations. AOA's in excess of 50° were achieved with aft cg when full aft stick was sustained while AOA's were usually obtained below 30° with nominal cg. The pitching moment due to sideslip was similar to that of the F-5E. Trim AOA with full aft stick was increased by four degrees over that of the F-5E due to the increased stabilator deflection (20° compared to 17°) and a slight reduction in longitudinal stability. The higher AOA's (as compared to the F-5E) obtained with sustained full aft stick

with an aft cg resulted partially from the increased stabilator deflection but primarily from the decreased longitudinal stability above 40° AOA. Abrupt full aft stick applications were capable of achieving in excess of 40 to 50° AOA with the nominal and aft cg's, respectively, without sustaining full aft stick (discussed later).

Lateral-Directional

The onset of wing rock at stall AOA was a result of static directional stability C_{n_β} becoming negative while maintaining sufficient dihedral effect C_{l_β} to prevent a pure nose slice from occurring. Excursions in sideslip were self-terminating as the airplane rolled due to C_{l_β} and thus reduced sideslip through the interchange of AOA and sideslip. Dihedral effect was, however, reduced somewhat near stall AOA, especially with flaps UP. Above approximately 28° AOA, strong dihedral effect was restored while static directional stability generally remained negative. Flap deflection increased dihedral effect for AOA's up to approximately 35°. The static directional instability was of greater magnitude than that of the F-5E, but an increase in dihedral effect due to the installation of wing fences (Figure C1) resulted in improved lateral-directional characteristics over that of the F-5E for AOA's below approximately 32°. Although airplane response to aileron was sluggish near or above stall AOA, no strong adverse effects were noted when full aileron was applied at high AOA. Aileron effectiveness $C_{l_{\delta a}}$ existed near or above stall AOA while $C_{n_{\delta a}}$ decreased significantly as AOA was increased to stall and therafter retained near constant effectiveness at AOA's above stall. Negligible yawing moment due to aileron $C_{n_{\delta a}}$ existed near or above stall AOA while C_n was small or negative. The result of aileron input was that the airplane initially rolled (essentially about the body X-axis) in the direction of input and built up adverse sideslip due to an interchange of AOA and sideslip. As stall AOA was approached, attainable roll rates with the ailerons were greatly reduced due to the dihedral effect associated with this adverse sideslip and decreased aileron effectiveness. At or above stall AOA, the adverse sideslip tended to cause the roll to hesitate or even reverse direction. Rudder became the primary roll

control near stall AOA. Rudder effectiveness $C_{n_{\delta r}}$ did not noticeably decrease until stall AOA was exceeded. Little, if any, effectiveness remained above 50° AOA. Rudder rolls near but below stall AOA were rapid and smooth. During these rolls, typical peak yaw and roll rates of 30 and 100° per second, respectively, caused AOA to peak to approximately 30 degrees even with fixed longitudinal stick position during the roll. As stall AOA was exceeded, the roll hesitated and became more oscillatory. At or above stall AOA, rudder inputs resulted in yaw excursions to quite large proverse sideslip (due to negative C_{n_β}). A rapid roll due to dihedral effect followed. As the airplane rolled, sideslip changed sign due to an interchange of AOA and sideslip. Thus, adverse sideslip was created, causing a rolling moment (due to C_{l_β}) opposite to the direction of control input. Depending on the magnitude of the adverse sideslip, the roll rate momentarily decreased, stopped, or even changed sign before sideslip again became proverse. Yaw rate normally remained the direction of rudder input. Above stall AOA the roll could be described as a continual rotation about the stability axis with roll oscillations (wing rock) superimposed. Examples of rudder rolls are presented in Figures A13 through A15. No problems existed in performing full rudder pedal rolls when longitudinal stick was maintained forward of or at that position required to trim the airplane at stall AOA. When full aft stick was applied (with aft cg) in conjunction with full rudder, however, peak AOA's in excess of 45° were obtained due to the stabilator deflection, inertial coupling, and pitching moment due to sideslip. With the attainment of these extreme AOA's, PSG or spin entry was possible (discussed later).

Significant yawing moments, observed in flight test data, were present at zero or small sideslip angles (less than + 10°) above approximately 54° AOA. These moments were presumably the result of the asymmetric shedding of vortices from the nose of the airplane (determined during ongoing wind tunnel and water tunnel flow visualization tests by the contractor). Above 42° AOA was significantly smaller. Flight test data indicated these yawing moments to be positive from approximately 45 to 60° AOA and negative between approximately 60 and 70° AOA. The sign of those below 45° AOA was not consistent. The yawing moments above 42° AOA were approximately twice the the magnitude of those of the F-5E.

Whereas these moments were not significant for the F-5E, they were very influential in the behavior of the F-5F. The existence of the large yawing moments above 42° AOA was one of the two primary differences between the F-5F and F-5E which resulted in decreased PSG and spin resistance (the other being decreased longitudinal stability above 40° AOA, discussed later). A propelling yaw damping derivative (positive Cnr) were evident above 50° AOA, with the strongest effect between 50 and 60° AOA. Significant effects due to positive C_n were evident when approximately 35° per second yaw rate was achieved in the 50 to 60° AOA region. The propelling yaw damping was similar to that of the F-5E.

PSG and Spin

The attainment of approximately 45° AOA by abrupt aft stick application alone was sufficient for PSG or spin entry. High AOA obtained in such a manner was initially accompanied by very small sideslip angles. Large yawing moments were present (discussed earlier) to establish a yaw excursion which was capable of causing PSG or spin entry. Figure A29 presents an example of an abrupt pullup at 150 KIAS with an aft cg which resulted in an unrecoverable flat spin. Although not pursued further, trends in data indicate certain conclusions about abrupt full aft stick inputs applied below stall AOA. With an aft cg in a one-g, wings level condition, abrupt full aft stick applied at 20° AOA or lower (greater than 130 KIAS) and sustained for as little as two seconds can result in an unrecoverable flat spin. With the nominal cg, such an input would be required at approximately 10° AOA or lower (greater than 160 KIAS) for spin entry to occur. Entry into a PSG or recoverable oscillatory spin, rather than a flat spin, may be possible with the nominal cg due to the increased nosedown aerodynamic pitching moment which could prevent the rapid transition to a flat spin by limiting AOA excursions. To achieve sufficient AOA for PSG or spin entry from an accelerated flight condition, the abrupt aft stick input must occur at significantly lower AOA. Above 250 KIAS, for instance, the input must occur at an AOA at least 5° lower than for the one-g, wings-level condition. Pitch damping C_{m_q} was the reason for the lower AOA requirement for full aft stick input during accelerated flight. More horizontal horizontal tail was required to obtain a given AOA, less incremental horizontal tail input was available to

increase AOA during an abrupt full aft stick input. In addition, the higher attainable pitch rates at the higher airspeeds produced more nosedown moment due to C_{m_q} than at low airspeeds. Full aft stick/full rudder or sustained full aft stick (smooth input) maneuvers achieved 45° AOA or higher with large sideslip oscillations (approximately + 20°). The airplane was susceptible as with the abrupt full aft stick input alone. The large yawing moments at small sideslip angles did not completely dominate the motion in these maneuvers since less time was spent at +10° sideslip than during the abrupt pullup maneuver. With the large sideslip oscillations, the natural "stability" (due to strong C_{l_β}) often contained the yaw excursions. However, the effect of the large moments at small sideslip was evident and resulted in either decreasing or increasing the existing yaw rates or starting a yaw rate if none existed. Thus, any maneuver which achieved near 45° AOA had the potential for PSG or spin entry. PSG or spin entry was possible without significant influence from the large yawing moments at small sideslip. If full aft stick/full rudder maneuvers were performed so as to achieve near 50° AOA with significant yaw rate (at least 25° per second), PSG entry was possible. Spin entry was possible if at least 35° per second yaw rate was established near 50° AOA. However, many full aft stick, full rudder deflection maneuvers (with or without cross controlled aileron) resulted in high yaw rate with lower AOA (less than 40°) or high AOA (greater than 50°) with low yaw rate but did not sustain both high AOA and high yaw rate. Two effects which often prevented a sustained high AOA were the nominal cg and high airspeed. The increased longitudinal stability at the nominal cg made it very difficult to sustain AOA's much above 40°. Maneuvers which maintained a nose low attitude, such as windup turns, maintained high enough speeds so that a substantial aerodynamic nosedown moment was present to counter the noseup inertial moment and thus prevent sustained at the extreme AOA's because of inertial yaw coupling (I_x - I_y/I_z pq). Yaw rates in excess of 30° per second with MANEUVER flaps and in excess of 40° per second with flaps UP were obtained below 40° AOA. The higher attainable yaw rates with flaps UP was due to less dihedral effect below approximately 35° AOA than with MANEUVER flaps. Therefore, the airplane was more susceptible to PSG/spin entry with flaps UP. The large positive pitch rates, involved in achieving even higher AOA, coupled with roll rate to produce an inertial yaw acceleration to

oppose the established yaw rate. This opposition to yaw rate, along with decreased rudder effectiveness at high AOA's, resulted in reduced yaw rates of less than 20° per second at the high AOA's. As a result, the following was the most critical full rudder/full aft stick maneuver from the standpoint of susceptibility to PSG or spin: a level, decelerating turn applying full rudder out of the turn and smooth full aft stick below stall AOA. Susceptibility was significantly increased when this maneuver was flown with the aft cg configured airplane or with flaps UP. Full rudder input below stall AOA produced sufficient yaw/roll rates to inertially couple AOA to large values. As the airplane pitched to high AOA/high pitch attitude (since rudder was applied away from the turn direction), the airspeed decreased rapidly. Thus, the aerodynamic nosedown moment was reduced and high AOA was more easily sustained. Smooth aft stick application (as opposed to abrupt) allowed the high AOA to be obtained with minimum pitch rate and thus minimum anti-spin inertial yaw acceleration. This maneuver established the high AOA while maintaining sufficient yaw rate to generate a substantial prospin yawing moment due to positive C_{n_β}. Therefore, a tendency towards continued rotation was established. Forward stick was the key to recovery from the PSG. Considerable nosedown moment due to the stabilator was required to overcome both the noseup inertial moment due to the yaw and roll rates and noseup moment due to side- slip. Depending on the established yaw and roll rates, full forward stick could be required for recovery from the PSG. It was possible to accelerate the yaw rate somewhat with the application of abrupt forward stick. A negative pitch rate (or reduction in positive pitch rate), produced by the forward stick, coupled with roll rate to create a prospin inertial yaw acceleration (or reduced the typical anti-spin inertial yaw acceleration by reducing positive pitch rate). If sufficient forward stick was not applied to simultaneously reduce AOA, progression into a developed spin was probable. In some cases, especially with the aft cg, full forward stick applied immediately upon recognition of loss of control did not effect recovery without entry into a spin.

When recovery was not effected from a PSG with the nominal cg, the propelling C_{n_r} increased yaw rate and an oscillatory spin was established. Full forward stick usually

did not produce sufficient nosedown moment to reduce AOA due to the increased inertial noseup moment as yaw rate increased. Reduction in yaw rate to reduce the noseup inertial moment, while maintaining forward stick, was the key to recovery from the spin. Forward stick was maintained to allow maximum nosedown aerodynamic moment. However, effectiveness of the lateral-directional controls to reduce yaw rate was marginal at best. Rudder applied against the spin produced little, if any, yawing moment to slow the rotation due to loss of effectiveness at high AOA. Ailerons applied in the direction of spin produced little, if any, adverse yawing moment to oppose the rotation since yawing moment due to aileron was minimal at high AOA. Probably the most benefit of the lateral-directional spin recovery controls was the rolling moment of the aileron. A very small roll capability into the spin direction caused a slight wing-down orientation about the spin axis (inside wing down). This produced a small positive pitch rate which coupled with the rollrate to produce an anti-spin inertial yaw acceleration. Recovery or non-recovery from the spin (above 50° AOA) was a result of the balance between this inertial anti-spin yaw acceleration and the prospin aerodynamic yaw acceleration primarily due to the positive C_{n_r}. AOA oscillations to lower than 50° resulted in an anti-spin aerodynamic yaw acceleration due to negative C_{n_r} and some rudder effectiveness, and resulted in a decrease in yaw rate. When yaw rate was reduced enough to allow a sustained AOA below 50°, recovery was accomplished by sustaining anti-spin controls. However, if AOA was sustained between 50 and 60° (region of strongest positive C_{n_r}) long enough, yaw rate could accelerate significantly, and progression into the higher rate, higher AOA unrecoverable flat spin could probably occur. With the aft cg, if recovery was not effected from a PSG, rapid transition into an unrecoverable flat spin occurred. This was due to the significantly reduced nosedown restoring pitching moment with the aft cg.

Centerline Tank Loading

Addition of a centerline tank (loading 5) to the basic airplane resulted in a significant degradation in static lateral-directional stability (Figures B11 and B12). Static directional

only slightly for higher AOA's. Dihedral effect was significantly reduced as compared to the basic airplane for all AOA's above approximately 10°. With flaps fully extended, C_{l_β} was reduced to near zero in a small AOA region, near 22 to 24 20. This AOA region was slightly larger, approximately 20 to 25° AOA, with minimal flap extension (as with MANEUVER flaps selected above 250 KIAS) or with flaps UP. With MANEUVER flaps, stalls resulted in a nose slice tendency when stall AOA was attained. The motion following the nose slice consisted of wing rock with more yaw rate and larger sideslip angles than were evident with the basic airplane. Above 200 KIAS and especially above 250 KIAS (only 12/8° LE/TE flaps), stalls resulted in a severe, abrupt nose slice at approximately 20° AOA. The nose translated purely in yaw until dihedral effect was restored at approximately 10° of sideslip. Restoration of dihedral effect was abrupt, resulting in large initial roll rate excursions. Large yaw rates were then combined with the large roll rates in the sane direction and AOA was inertially coupled abruptly to over 60°. The roll following the nose slice resulted in a large sideslip buildup to the opposite direction due to an interchange of AOA and sideslip. The directional instability was such that this sideslip caused either a reduction in established yaw rate or a yaw and roll opposite in direction to the nose slice. This abrupt yaw and roll resulted in a very large peak in inertial pitch acceleration, causing a peak AOA in excess of 75° during the PSG. Figures A9 and A26 present the maneuvers performed during this program with the centerline tank loading.

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